CHAPTER 2. PART 29
AIRWORTHINESS STANDARDS
TRANSPORT CATEGORY ROTORCRAFT

SUBPART F - EQUIPMENT

EQUIPMENT - GENERAL

AC 29.1301. § 29.1301 FUNCTION AND INSTALLATION.

a. Explanation. It should be emphasized that this rule applies to each item of installed equipment which includes optional equipment as well as required equipment.

b. Procedures.

(1) Information regarding installation limitations and proper functioning is normally available from the equipment manufacturers in their installation and operations manuals. In addition, some other paragraphs in this AC include criteria for evaluating proper functioning of particular systems. (An example is paragraph AC 29 MG 1 for avionic equipment.)

(2) This general rule is quite specific in that it applies to each item of installed equipment. It should be emphasized, however, that even though a general rule is relevant, a rule that gives specific functional requirements for a particular system will prevail over a general rule. Therefore, if a rule exists that defines specific system functioning requirements, its provisions should be used to evaluate the acceptability of the installed system and not the provisions of this general rule. It should also be understood that an interpretation of a general rule should not be used to lessen or increase the requirements of a specific rule. Section 29.1309 is another example of a general rule, and this discussion is appropriate when applying its provisions.

AC 29.1303. § 29.1303 (Amendment 29-24) FLIGHT AND NAVIGATION INSTRUMENTS.

a. Explanation. This rule lists the flight and navigation instruments that are required for VFR certification. Several additional rules to be consulted when determining the flight and navigation instrument installation design are § 29.1321, arrangement and visibility, § 29.1331, flight instrument power supplies, and § 29.1333, systems that operate the flight instruments at each pilot station. Additional information regarding the different instruments can be found by referring to the Technical Standard Order (TSO) for each one. Compliance with the TSO requirements does not ensure compliance with the appropriate Part 29 requirements; however, satisfactory installation normally results. Other considerations may also be found by reviewing the requirements of §§ 29.1323, 29.1327, 29.1335, 29.1381, 29.1543, 29.1545, 29.1547, and Part 29, Appendix B. Paragraphs VIII(a) and (b) of Appendix B include IFR operation considerations for flight and navigation instruments. In addition, if the maximum allowable airspeed is dependent on conditions such as weight and altitude, that information is normally provided on tables, graphs, placards, or other means in the cockpit and in the rotorcraft flight manual.

b. Procedures.

(1) The following instruments are considered to be flight instruments:

(i) Airspeed indicator.

(ii) Sensitive altimeter.

(iii) Free-air temperature indicator.

(iv) Nontumbling gyroscopic bank and pitch indicator.

(v) Gyroscopic rate-of-turn indicator with an integral slip-skid indicator.

(vi) Rate-of-climb (vertical speed) indicator.

(2) The remaining instruments are navigation instruments.

(3) If a speed warning device is required to be included as part of the rotorcraft design, it must meet the performance requirements given in the rule. In addition, the evaluation of the acceptability of the aural warning (i.e., this warning differs distinctively from aural warnings used for other purposes) should be accomplished by flight test personnel as part of their overall cockpit evaluation.

(4) Electronic Flight Instrument Systems (EFIS).

(i) Explanation. The increased use of microprocessor technology in avionic systems has resulted in the use of computer-generated graphics to replace conventional electromechanical instruments which are used for the display of flight information required by § 29.1303(f), (g), and (h). For IFR certified aircraft, the EFIS usually is used for the display of the magnetic gyro-stabilized direction indicator (slaved compass system.) These computer-generated graphics are usually displayed on small multicolor-shadow-mask cathode ray tubes (CRT) and replace the horizontal situation indicator (HSI) and the attitude direction indicator (ADI). This discussion presumes that the EFIS for which approval is sought meets the general requirements of an EFIS for a transport category airplane with regard to color, symbology, operation, and so forth. This paragraph along with some others in this document principally highlights the areas which are peculiar to the installation in a transport category rotorcraft. A discussion of the flight director function performed by the EFIS is given in paragraph AC 29.1335. A discussion of the location of the displays is contained in paragraph AC 29.1321. A discussion of the requirements for an EFIS in a transport category airplane is contained in AC 25-11, Transport Category Airplane Electronic Display Systems, dated September 16, 1987.

(ii) Procedures.

(A) System Components. The system components require qualification testing to determine that their design is acceptable, free from hazards, and suitable to their airborne environment. Generally the components of the EFIS should meet the requirements of TSO C-113.

(1) Environmental Qualification. The EFIS hardware must be shown to be suitable to its airborne environment. A desirable way to qualify the system component is to obtain approval to the appropriate TSO. If the equipment is not TSO approved, it should be shown via testing that it complies with the requirements of SAE Document AS-8034. This will include testing in accordance with the appropriate categories of the latest revision of RTCA Document DO-160, JEDEC Publication No. 64D (Protection from Ionizing Radiation), and UL Document No. 1418 (Impact Implosion Test).

(2) Software. The embedded software should be qualified to an appropriate standard. The software level is contingent on the worst case criticality of the function it performs. As an example, the display of incorrect roll and/or pitch by an EFIS ADI instrument is a hazardous or “critical” malfunction. The software should be designed to provide adequate consideration for this factor (reference paragraph AC 29.1309). A similar consideration is required for altitude and airspeed.

(B) System Installation Consideration.

(1) Display Chromaticity and Luminance. The chromaticity and luminance of the displays should be determined to be acceptable for all critical cockpit lighting conditions which are expected in service. An expanded discussion of these characteristics may be found in AC 25-11.

(2) Temperature Survey to Determine Proper Cooling of EFIS Components.

(i) Equipment Requiring Cooling Test. As with any avionic equipment, good engineering judgment may deem that all components of the EFIS should have an in-flight temperature survey performed to ascertain that the thermal environmental tolerance of the system components is not exceeded. Usually, the following general guidelines may be used to aid in determining when an in-flight temperature survey is warranted.

(a) Components which contain a CRT require the temperature survey outlined in paragraph AC 29.1303b(4)(ii)(B)(1)(ii).

(b) Equipment which does not contain a CRT, but is specified by the manufacturer to require forced air cooling (by an airframe mounted system), usually requires a temperature survey.

(c) Equipment which does not contain a CRT and is not specified as requiring forced air cooling may usually have its critical thermal environment substantiated by laboratory testing.

(ii) Temperature Survey Testing. The temperature tests for the EFIS units should consist of a short-term test of approximately 30 minutes which accounts for an aircraft which has heat-soaked on the ramp. A factor of 25° F should be added to the maximum corrected temperature to account for “greenhouse effect.” A long-term test should be accomplished at various altitudes and limiting (low and high) airspeeds. All avionic equipment should be turned on during this test, and the cockpit panel lights should be operated at full intensity. The environmental control unit (ECU) or air-conditioning system should not be operating during these tests; however, any windows or vents which are part of the “basic” TC rotorcraft may be utilized to ventilate the pilot’s stations. Both these tests should be corrected to the maximum temperature for which the rotorcraft is certified and a standard lapse rate for altitude as specified in this AC. If an airframe cooling system is necessary to keep the display units within acceptable temperature limits, then the pilot(s) must be made aware of a failure or malfunction of this cooling system. Some type of cockpit visual annunciation with the capability to perform a preflight test is usually utilized to fulfill this requirement.

(3) System Reliability.

(i) Failure of the EFIS to perform a required function which results in the reversion to standby instruments or requires the use of abnormal procedures should be shown to be improbable (for Category A rotorcraft, reference § 29.1309(b)).

(ii) For IFR operations, Appendix B of Part 29, paragraph VIII(b)(5)(ii), requires that the equipment, systems, and installations must be designed so that one display of the information essential to the safety of flight which is provided by the instruments will remain available to a pilot, without additional crewmember action, after any single failure or combination of failures that is not shown to be extremely improbable. The display of attitude, altitude, or airspeed is individually “essential to the safety of flight,” and, therefore, the loss of all attitude display, all airspeed information, or all altitude information to the pilot(s) should be extremely improbable. Also, any malfunction or failure of the EFIS which would result in the display of simultaneous incorrect display of this critical information should be extremely improbable. In view of the relatively new technology embodied in the EFIS, the conventional technology electromechanical standby attitude indicator, with its independent power supply, should be retained.

(4) Symbology and Function. When assessing the acceptability of the EFIS, consideration should be given to the effect of the loss of one of the CRT color

guns. This type of failure is especially a factor in determining the acceptability of the installation for single-pilot operation.

(5) Circuit Protective Devices. All circuit protective devices for the EFIS and related, required interface units should be placed in the cockpit where they can be reset by one of the pilots without leaving the pilot seat.

c. Standby Instruments. The EFIS which have been approved on transport category rotorcraft at this time have only presented the critical function of attitude display. A specific requirement for a standby attitude instrument is contained in Appendix B of Part 29. This requirement is usually satisfied by an electromechanical panel-mounted gyro with an independent power source. This type of installation is what was envisioned by the authors of Appendix B. Because of the mature technology of this type of standby attitude indicator, certain aspects of the EFIS installation have not been an area for concern. If, however, a total commitment of critical display functions is made to the "glass" technology, such that the standby attitude instrument requirement is satisfied by a software based CRT system, then several major areas of concern will be raised. Among these are the electromagnetic vulnerability of the system (protection from the effects of lightning and high energy radio frequency fields) and software. The certifications of EFIS with an electromechanical standby attitude indicator have not considered loss of function critical from a software aspect. (reference paragraph AC 29.1309 for a discussion of software qualification.)

AC 29.1305. § 29.1305 (Amendment 29-10) POWERPLANT INSTRUMENTS.

a. Explanation. This section specifies those instruments which are required for reciprocating and turbine engine installations. It also provides instrumentation requirements for operating rotorcraft in Category A or Category B. These instruments will provide the pilot with essential data to determine operational status of critical components and select desired performance conditions.

b. FAR 29.1305(a)(4), (a)(6), and (a)(9) requires a warning device for low fuel, gearbox oil pressure, and transmission oil temperature. An indicator/gage is not acceptable for use as a warning device since the indicator/gage is not a primary instrument and therefore is not actively monitored.

c. There are advanced display systems that take advantage of microprocessor power by integrating the processing of several parameters. These systems have to date been referred to as Engine Caution Advisory Systems (ECAS) or as Integrated Instrument Display Systems (IIDS) and possibly other variations of these names. These systems typically integrate propulsion instruments, fuel quantity indication, and caution and warning system into a single display system. In traditional designs the powerplant instruments, fuel quantity display, and the caution and warning system are independent from each other. The integration of these systems/indicators eliminates their independence from one another and increases the probability of loss of more than one indicator/system as a result of a single fault or malfunction. Redundant design is generally applied to compensate for the loss of independence.

(1) This integration and resultant mitigation of independence can result in an increased opportunity for common mode failures. Approval of the compensating features is elevated in importance as it is this aspect that allows the concept to be acceptable and subsequently certifiable. The loss of all displayed information or erroneous information should be considered for determination of worst case criticality. With this determination of criticality, the design can be evaluated to see that it meets the minimum associated level of design assurance. Additionally, due to space limitations, some systems employ “page over” features that may have some difficulty displaying the required information when needed and human factors aspects must be considered.

(2) The instrument display system must be investigated and found to be acceptable under both normal and emergency conditions, must perform its intended function under foreseeable operating conditions, and must be designed to minimize the hazards in the event of probable malfunction or failure.

(3) It must be shown that there is appropriate redundancy to provide adequate compensation for the loss of independence in the system. If a multi-page system is employed, it must be shown that needed information is displayed when required. Specific issues that must be addressed to assure compliance with the minimum safety standards are as follows:

(i) The level of most severe hazard must be determined.

(ii) Equivalent reliability and software design assurance to the determined criticality level must be shown.

(iii) Pretest capability must be provided for the warning and caution system to preclude an associated latent failure.

(iv) Human Factors (reference §§ 29.1321 and 29.1322).

d. Additional rules to be consulted when determining the powerplant instrument installation design are §§ 29.1321, Arrangement and visibility; 29.1337, Powerplant instruments; 29.1381, Instrument lights; 29.1543, Instrument markings: General; 29.1549, Powerplant instruments; 29.1551, Oil quantity indicator; and 29.1553, Fuel quantity indicator.

AC 29.1305A. <u>§ 29.1305 (Amendment 29-26) POWERPLANT INSTRUMENTS</u>.

a. <u>Explanation</u>. Amendment 29-26 revises, edits, and adds new powerplant instrument requirements. Section 29.1305(a)(4) was revised to require a low fuel warning device for each tank that can be used to feed an engine. The amendment

allows a longer time between warning actuation and fuel exhaustion and requires the low fuel warning device to be independent of the normal fuel quantity indication system. Section 29.1305(a)(17) was changed to extend its application to all rotorcraft (not just those with turbine engines) and to require an indication to the crew of the degree of filter blockage as it relates to the fuel flow requirements in § 29.955.
Section 29.1305(a)(19) was revised to require function indicators only for fuel heaters that can be selected or are controllable. A new paragraph (a)(20) was added to § 29.1305 that combined identical requirements for fuel pressure indicators in paragraphs (b)(2) and (c)(2) and modified the applicability of these requirements to only those fuel systems with devices or components that could adversely affect fuel pressure at the engine, if they fail. It also eliminates the requirement for fuel pressure indicators in fuel systems, such as suction or gravity feed systems, which do not incorporate pumps or filters. A new § 29.1305(a)(21) was added that requires a warning device to indicate to the flight crew the failure of any fuel pump that is required to supply adequate fuel flow to the engine according to § 29.955; such indication is not required for fuel pumps which are demonstrated to be only necessary for engine starting. Section 29.1305(a)(22) adds a requirement for a warning or caution device to alert the flight crew when particles are detected by the chip detector required by § 29.1337(d). A new § 29.1305(a)(23) added a requirement for powerplant instruments or warning devices for auxiliary power units installed in rotorcraft.

b. Procedures. The requirement and purpose for each instrument is self-explanatory in the amendment. Other sections that should be considered when designing powerplant instruments are listed in paragraph AC 29.1305.

AC 29.1305B. 29.1305 (Amendment 29-34) POWERPLANT INSTRUMENTS.

a. Explanation. Amendment 29-34 added §§ 29.1305(a)(24) and 29.1305(a)(25) to provide for 30-second/2-minute OEI power ratings.

(1) Section 29.1305(a)(24) adds the requirement that a device or means be provided to alert the crew of the use of the 30-second and 2-minute OEI power level. The crew should be alerted when the 30-second or 2-minute interval begins and when the time interval ends. The amount of time spent at the 30-second or 2-minute OEI power levels is at the crew's discretion, unlike the other limits for 30-second OEI that are set by an automatic control required by § 29.1143. The purpose for providing the time interval alerts and automatically controlling the 30-second OEI limits is to free the crew from monitoring the engine instruments during critical phases of flight caused by the loss of an engine.

(2) Section 29.1305(a)(25) adds the requirement for a device to record the usage and the amount of time spent at the 30-second OEI power level and the amount of time spent at the 2-minute OEI power level. The information recorded by this device is for the use of the ground crew to determine if maintenance actions/inspections are to be conducted.

b. <u>Procedures</u>. For the purpose of complying with FAR 29.1305(a)(24) and 29.1305(a)(25), the 2-minute OEI power level is considered to be achieved whenever one or more of the operating limitations applicable to the next lower OEI power rating are exceeded. The 30-second OEI power level is considered to be achieved whenever one or more of the operating limitations applicable to the 2-minute OEI power rating are exceeded.

(1) A review of the method to meet the requirements of § 29.1305(a)(24) should be conducted by flight test personnel. A determination should be made as to whether the method used to alert the crew of 30-second or 2-minute OEI power usage can be recognized and understood by the crew.

(2) To meet the requirements of Section 29.1305(a)(25), a device should be installed on the engine or the airframe to record the time and each usage of 30-second and 2-minute OEI power levels. The information on the time and usage of 30-second and 2-minute OEI power should be recoverable from the recording device by ground personnel. The device should not be capable of being reset in flight and should only be capable of being reset by ground personnel. Prior to each flight this device should be capable of being checked for proper operation and to determine if 30-second or 2-minute OEI power levels were used during the previous flight.

c. <u>Integrated Display Systems</u>. This advisory material is to provide guidance for compliance to Part 27 and Part 29 regulations as they apply to integrated display systems. The integration aspects of these systems require some additional issues to be addressed during certification. The term “must” in this advisory material is used in the sense of ensuring the applicability of these particular methods of compliance when the acceptable means of compliance described herein is used. This advisory material establishes an acceptable means, but not the only means of certifying an integrated display system.

(1) <u>Definitions</u>.

(i) <u>Integrity</u>. The term “integrity” for the purpose of this advisory material includes the hardware quality requirements, including reliability; as well as the software level requirements, as defined in DO178B.

(ii) <u>Criticality</u>. The term “criticality” refers to the five levels of criticality addressed in FAA Advisory Circulars AC 27-1B and AC 29-2C.

(2) <u>Related documents</u>.

(i) Federal Aviation Regulations (FARs) paragraphs 21.21, 29.1301, 29.1309, 29.1305, and 29.1322

(ii) Standards - Latest revision of RTCA/DO178 and RTCA/DO-160; SAE documents

(iii) ARP4754 and ARP4761

(3) Background. A tendency to integrate functions/indications that have previously been independent is a result of technology advancement. Microprocessor driven systems have facilitated the ease of this integration. Integrated Instrument Display System (IIDS) or Engine Instrument and Caution Advisory System (EICAS) are examples of integrated display systems. IIDS, EICAS, or any other similar systems are defined as a combination of engine instruments (previously independent indicators), fuel quantity indication, and caution/warning parameters, as a minimum, presented by a common display driven by a common processor.

(4) Discussion. This design philosophy does not result in the traditional requirement for individual display independence for failure/malfunction considerations. This loss of independence means that a single failure could result in loss of most, if not all, instrument displays on the integrated display system. Redundancy of the integrated display system is often proposed to compensate for this lack of independence. However, redundancy alone may not meet the integrity requirements since they are derived from the level of criticality associated with the loss or malfunction of instrument/parameter displays for flight operations that are dependent on these indications.

(5) Certification Approach. A two step procedure should be used to determine the adequate safety level for this type of system. The first step is to determine the level of criticality associated with the total loss/malfunction of these functions/indications or loss/malfunction of the critical parts of the display. This can be achieved through the use of a functional hazard assessment (FHA). This criticality assessment must be a product of failure/malfunction of the indication system and the flight operation that would represent the worst case for loss of this information. The second step is to determine that the design integrity of the system is at least equal to the assessed criticality level determined in step one.

(6) Functional Hazard Assessment. The operational classifications to be considered when assessing the criticality are Cat A, Cat B, and IFR. The need for critical information varies with each of these different operational categories. An example would be the demand for OEI parameter information in the single engine Cat A operation. Another example is the loss of fuel quantity indication and fuel low level indication simultaneously in IFR flight conditions. The FHA should address not only loss of one type of indication, but combined loss of engine parameter indication, including total loss of display information, caution/warning, fuel quantity indication, and any other included display in combination with a particular flight operation. There are techniques to lessen the consequences of the failure/malfunction requirements for integrity, such as providing back-up displays for the information deemed critical for a particular operational consideration.

(7) Summary. The loss of all integrated display information for certain types of flight operations may have the highest level of criticality associated with it. The same may be true for malfunctions that result in misleading indications. These failures/malfunctions must be addressed by the commensurate design integrity level. Lesser levels of criticality must also be addressed by the appropriate design integrity levels.

AC 29.1305C. § 29.1305 (Amendment 29-40) POWERPLANT INSTRUMENTS.

a. Explanation. Amendment 29-40 added section 29.1305(a)(6) to require an oil pressure indicator for each pressure-lubricated gearbox. Paragraphs (a)(6) through (a)(25), prior to this amendment, have been redesignated as paragraphs (a)(7) through (a)(26).

b. Procedures. In addition to providing an oil pressure indicator for each pressure-lubricated gearbox, the guidance material of the previous AC 29.1305 paragraphs continues to apply.

AC 29.1307. § 29.1307 (Amendment 29-12) MISCELLANEOUS EQUIPMENT.

a. Explanation. This rule provides a listing of several items of required miscellaneous equipment. Each item seems to be self-explanatory except the one requiring a master switch arrangement for electrical circuits other than ignition. The purpose of a master switch arrangement is to allow rapid removal of all bus loads from sources of electrical power in an emergency situation.

b. Procedures. When reviewing possible solutions to the master switch arrangement requirement, the following considerations should be included.

(1) System separation. Since wiring from each electrical system will be brought in close proximity to each other, extra care should be taken to maintain some separation. As examples, common connectors, common grounds, and common wire routing should be avoided.

(2) Installation of switches. The single switch should be avoided since it introduces the possibility of a single failure turning off the entire electrical system. One solution that is commonly used provides a close grouping of the switches such that the pilot can easily reach all switches and turn them all off with one action. This solution requires a cockpit evaluation to ensure the installation will be suitable for different hand sizes. Another solution involves a gang bar that can be moved with a single motion to turn off all sources. This solution has been found to be acceptable in several instances. Other solutions should be evaluated on their own merits, and the primary emphasis should be on maintaining some minimum system separation and conducting a cockpit evaluation by flight test personnel.

AC 29.1309. § 29.1309 (Amendment 29-40) EQUIPMENT, SYSTEMS, AND INSTALLATIONS.

a. Types of Equipment. This regulation covers, but is not limited to electrical, pneumatic, and hydraulic power sources, associated distribution, and corresponding utilization systems.

b. Environmental Qualification.

(1) Laboratory Tests.

(i) Environmental Standards. In order to assure that the components/systems under consideration will function properly when exposed to adverse environments, they should be tested in the laboratory under a simulated adverse environment. If a TSO exists and it is appropriate in environmental range and performance for an equipment installation, it is preferable the equipment be TSO approved. If there is no applicable TSO or an existing TSO does not provide for a sufficiently adverse environment, the latest revision of the Radio Technical Commission for Aeronautics (RTCA) document DO-160 is an acceptable environmental standard for laboratory qualification of aircraft equipment.

(ii) Adverse environmental variables for all types of required and critical equipment include, but are not necessarily limited to temperature, humidity, vibration, shock, altitude, overpressure, and power source transients.

(iii) For electrical/electronic equipment, adverse environmental variables include all of (b) above plus overvoltage and undervoltage. Electronic equipment should also be tested for electromagnetic interference (EMI). These tests should include both emission and susceptibility evaluations of both conducted and radiated EMI.

(iv) Explosion Tests. Those items of electrical/electronic equipment that are to be located in areas subject to flammable fluids and vapors, as a result of any single probable malfunction or failure, including failure of couplings or lines should be tested as an ignition source. These tests consist of normal operation of the equipment in a physically contained explosive atmosphere. The explosion test procedure in the latest revision of DO-160 will satisfy this requirement. Paragraph AC 29.863 provides further guidance on safety from explosion. If another standard is used that is at least as good as the latest revision of DO-160, it may also be accepted to satisfy this requirement.

(2) Installed Environmental Tests. After the environmental ratings of the components/systems have been established, it should be assured that as installed, these ratings will not be exceeded. Normally, installed equipment need not be instrumented and tested in flight nor is it necessary to instrument the compartment or rack where the equipment is installed. Satisfactory environment and equipment

compatibility are assured by selection of the proper environmental category of laboratory tests. The category is determined by the type of aircraft (reciprocating or turbine) and flight envelope (altitude and temperature). Exceptions to normal installations are (a) Alternator/generator cooling, where radiated and conducted heat is almost always uncertain, also cooling air temperatures and flow rates are uncertain; (b) Where flight tests reveal excessive instrument panel vibration. In this case, the panel should be instrumented, tested, and, if necessary, design improvements made; and (c) Any other cases where good engineering judgment and application of sound engineering principles indicate a high likelihood that the installed environment is more severe than the equipment is capable of operating within.

(i) Temperature Tests.

(A) Temperature tests may be accomplished by instrumenting the installed equipment environment with a recorder that provides a permanent record of time, altitude, and temperature. The pertinent temperature should be recorded as the rotorcraft is operated throughout its altitude range, including ground operation. The maximum and minimum temperatures recorded should be corrected degree for degree to assure the equipment under test remains within its temperature rating while the rotorcraft operates throughout its approved ambient temperature envelope. (For generator/alternator cooling test procedures, refer to paragraph AC 29.1351.) Section 29.1043, paragraph (b) requires the maximum approved operating OAT to be at least 100° F for powerplant-mounted accessories such as starter generators, vacuum pumps, etc. Due to the impracticality of the 100° F hot day temperature limit, rotorcraft systems mounted on the powerplant are normally evaluated for at least 115° F hot day sea level conditions with corresponding 3.6° F/1,000-foot correction. The maximum hot day OAT at sea level must be specified in the rotorcraft flight manual. Section 29.1043, paragraph (b) is the regulatory basis for the lapse rate of 3.6° F/1,000 feet. This lapse rate should be applied regardless of the hot day sea level temperature the applicant chooses to certify for operation.

(B) The § 29.1043 paragraph (b) maximum ambient temperature definition should not be confused with operating temperatures in closed areas. Closed equipment rack areas can easily reach temperatures of 140° F when sitting on the ramp in the southern United States in midsummer. Normally, proper selection of the altitude temperature category in the latest revision of DO-160 will assure compliance.

(C) In some cases, the equipment manufacturer furnishes temperature limits for internal critical parts. For example, brushes, bearings, or field windings on DC generators. In these cases it is better to record the critical component temperature rather than equipment or equipment environment temperature.

(D) The following will illustrate an acceptable high temperature evaluation method:

$T_{OAT\ MAX}$ = Maximum outside air temperature at which temperature tests are conducted.

T_{MAX} = Maximum temperature to which the installed equipment has been tested in the laboratory.

$T_{TEST\ MAX}$ = Maximum installed equipment temperature recorded during tests.

T_{ORH} = The high reference outside air temperature. It varies with altitude starting at the highest sea level temperature at which rotorcraft operation is to be approved and decreases at 3.6° F/1,000 foot altitude. It can be no less than 100° F (reference § 29.1043(b)); however, it can be as high as the applicant wants.

$T_{H\ MAR}$ = Temperature margin between the maximum equipment temperature substantiated in the laboratory and the maximum installed equipment temperature when the rotorcraft is operating in the highest available OAT and approximately corrected at the altitude under consideration. If the margin is zero or positive, the equipment passes. If the margin is negative, the equipment fails the test.

$T_{H\ MAR} = T_{MAX} - (T_{TEST\ MAX} + (T_{ORH} - T_{OAT\ MAX}))$

<u>Example #1</u>: Assume the applicant is seeking approval for rotorcraft operation at the lowest acceptable OAT, at sea level, of 100° F and T_{MAX} for Generator Brush = 295° F at maximum load current throughout the altitude range. In-flight test data are:

Altitude (ft, MSL)	Measured Brush Temp ($T_{TEST\ MAX}$)	OAT = $T_{OAT\ MAX}$
sea level	275° F	90° F
5,000	270° F	80° F
10,000	285° F	60° F
15,000	294° F	42° F
20,000	290° F	20° F

First, T_{ORH} must be calculated for each altitude test point.

@ sea level, T_{ORH} = 100° F
@ 5,000 ft., T_{ORH} = 100° F -5,000 ft., x 3.6°/1,000 ft. = 82° F
@ 10,000 ft., T_{ORH} = 100° F - 10,000 ft. x 3.6°/1,000 ft. = 64° F
@ 15,000 ft., T_{ORH} = 100° F - 15,000 ft. x 3.6°/1,000 ft. = 46° F
@ 20,000 ft., T_{ORH} = 100° F - 20,000 ft. x 3.6°/1,000 ft. = 28° F

Then at sea level:

$T_{H\ MAR}$ = 295 - (275 + (100 - 90))

At 5,000 feet:

$T_{H\ MAR}$ = 295 - (270 + (82 - 80)) = 23° F

At 10,000 feet:

$T_{H\ MAR}$ = 295 - (285 +(64 - 60)) = 6° F

At 15,000 feet:

$T_{H\ MAR}$ = 295 - (294 + (46 - 42)) = -3° F

At 20,000 feet:

$T_{H\ MAR}$ = 295 - (290 + (28 -20)) = -3° F

Since $T_{H\ MAR}$ comes out negative at the 15,000- and 20,000-foot points, the generator fails. It will be necessary for the applicant to reduce the maximum load current, improve cooling, or otherwise change the design to assure the generator is operating within its approved temperature limit of 295° F.

(E) In most cases, the equipment is laboratory tested to minimum temperatures as severe as that of the rotorcraft's maximum certified altitude on a minimum temperature day. Therefore, unless equipment minimum temperature is affected by refrigeration or other temperature reducing environments, actual installed instrumented minimum temperature tests are unnecessary. If low temperature evaluation is necessary for the installed equipment, the following is an acceptable method:

$T_{OAT\ MIN}$ = Minimum outside air temperature at which temperature tests are conducted.

T_{MIN} = Minimum temperature to which the installed equipment has been tested in the laboratory.

$T_{TEST\ MIN}$ = Minimum installed equipment temperature recorded during tests.

T_{ORL} = The low reference outside air temperature. It varies with altitude starting at the lowest sea level temperature at which rotorcraft operation is to be approved and decreases at 3.6° F/1,000-foot altitude.

$T_{1\ MAR}$ = Temperature margin between the minimum equipment temperature substantiated in the laboratory and the minimum installed equipment temperature. If the margin is zero or positive, the equipment passes. If the margin is negative, the equipment fails the test.

$$T_{1\ MAR} = -(T_{MIN} - (T_{TEST\ MIN} + (T_{ORL} - T_{OAT\ MIN})))$$

NOTE: This equation assumes all temperatures are negative. It is necessary to place a (-) in front of the right side of the equation in order to convert the $T_{1\ MAR}$ value to the conventional positive answer for acceptance and a negative answer for rejection. Temperature in the 0 to 32° F range can be handled by conversion to the centigrade scale.

Example #2: Assume the applicant is seeking a low temperature operating limit at sea level of -25° F. Assume the hydraulic control cylinder has been substantiated in the laboratory to operate at a cylinder temperature of -40° F. The in-flight test data are:

Altitude (ft, MSL)	Measured Cylinder Temp ($T_{TEST\ MIN}$)	OAT = $T_{OAT\ MAX}$
sea level	0° F	-25° F
5,000	-9° F	-45° F
10,000	-21° F	-59° F
15,000	-32° F	-65° F
20,000	-40° F	-69° F

(T_{orl} must be calculated for each altitude test point)

@ sea level, T_{ORL} = -25° F
@ 5,000 ft., T_{ORL} = -25° F -5,000 ft., x 3.6°/1,000 ft. = -43° F
@ 10,000 ft., T_{ORL} = -25° F - 10,000 ft. x 3.6°/1,000 ft. = -61° F
@ 15,000 ft., T_{ORL} = -25° F - 15,000 ft. x 3.6°/1,000 ft. = -79° F*
@ 20,000 ft., T_{ORL} = -25° F - 20,000 ft. x 3.6°/1,000 ft. = -97° F*

*According to § 29.1043(b), the lowest temperature to be considered is -69.7° F.

Then at sea level:

$T_{1\ MAR}$ = -(-40 -(0 + (-25 -(-25)))) = 40° F

At 5,000 feet:

$T_{1\ MAR}$ = -(-40 - (-9 + (-43 - (-45)))) = 33° F

At 10,000 feet:

$T_{1\ MAR}$ = -(-40 - (-21 + (-61 - (-59)))) = 17° F

At 15,000 feet:

$T_{1\ MAR}$ = -(-40 - (-32 + (-69.7 - (-65)))) = 3.3° F

At 20,000 feet:

$T_{1\ MAR}$ = -(-40 - (-40+ (-69.7 - (-69)))) = -0.7° F

It can be seen that there is an acceptable margin at all altitudes up to and including 15,000 feet. However, at 20,000 feet, the margin is negative and the system fails.

(ii) Vibration tests. Normally, installed vibration tests are not necessary for equipment qualified in accordance with the latest revision of RTCA document DO-160. This paper categorizes vibration tests according to installed rotorcraft equipment location such as fuselage, engine compartment, instrument panel, equipment rack, etc. However, installed equipment vibration tests may be necessary when it appears the equipment location environment may exceed the laboratory-tested equipment vibration limits.

(iii) Altitude tests. If the equipment has been laboratory tested to the maximum certified altitude of the rotorcraft, installed altitude tests are unnecessary. The installed equipment must be either laboratory tested or tested in the rotorcraft to the maximum certified altitude of the rotorcraft.

(3) Lightning Strike Protection of Full Authority Digital Engine Controls.

(i) Explanation.

(A) The following discussion is written specifically for full authority digital engine controls (FADEC) with an alternate technology backup fuel control installed on rotorcraft with Category A engine isolation. The requirement for increased consideration of lightning strike encounter effects on avionic equipment and systems has been brought about by the increased use of avionics to perform functions, the failure or malfunction of which could result in a hazard to the rotorcraft. The susceptibility of current high technology avionic systems is increased by the use of large scale integration (LSI), very large scale integration (VLSI), and complementary metallic oxide silicon (CMOS) technologies which exhibit a greatly reduced tolerance to large amplitude, low energy electrical transients as compared to conventional bipolar technology, and the reduced physical protection and electromagnetic shielding afforded aircraft avionic systems by the advanced technology composite airframe materials. Additionally, processor-based systems have the failure phenomenon of digital upset. A digital upset occurs when a system, perturbed by an electrical transient, ceases proper operation in accordance with its embedded software while suffering no apparent component or device damage.

(B) Since elements of electrical/electronic engine subsystems are typically spread throughout much of the rotorcraft, transients caused by lightning are coupled into the subsystem interface cables and may damage the system or cause upset. Effective lightning protection must be designed and incorporated into these systems. Reliance upon redundancy as a means of protection against lightning effects is generally not adequate because lightning electromagnetic fields and structural IR voltages usually interact (to some extent) with all electrical wiring aboard a rotorcraft.

(C) The testing and analysis outlined in this discussion are methods by which the FAA/AUTHORITY may be assured that when the rotorcraft experiences “the foreseeable operating condition” of a worst-case lightning strike encounter that the electronically controlled engines will continue to “perform their intended function” and therefore be in compliance with § 29.1309(a) as installed.

(D) The definition of what constitutes a full authority engine control is not at this time clearly defined. However, it has been accepted in past certification that any control which relies upon the electronics for the function on which Civil Certification or Military Qualification is based (e.g. rotor speed governing) is a full authority control, regardless of the backup control mode provided. If engine certification or qualification can be achieved without the electronic control which is subsequently added to achieve improved operational efficiency in the aircraft, the control is “supervisory.” However, if the controls are used in a multiengine rotorcraft, a common failure caused by a lightning strike could result in simultaneous failures which would cause a reduction in power greater than the loss of one engine. This would also be considered “full authority.”

NOTE: If OEI ratings are approved, cumulative loss of power from all engines should be limited to allow flight manual performance based on OEI ratings.

(ii) Procedures. Although not a regulatory requirement, it is recommended that a formal written certification plan be used to assure regulatory compliance. The use of this plan is beneficial to both the applicant and the FAA/AUTHORITY because it identifies and defines an acceptable resolution to the critical issues early in the certification process. These are the usual steps to be followed when utilizing a certification plan:

(A) Prepare a certification plan which describes the analytical procedures and/or the qualification tests to be utilized to demonstrate protection effectiveness. Test plans should describe the rotorcraft and FADEC system to be utilized, test drawing(s) as required, the method of installation that simulates the production installation, the lightning zone(s) applicable, the lightning simulation method(s), test voltage or current waveforms to be used, diagnostic methods, and the appropriate schedules and location(s) of proposed test(s).

(B) Obtain FAA/AUTHORITY concurrence that the certification plan is adequate.

(C) Obtain FAA/AUTHORITY detail part conformity of the test articles and installation conformity of applicable portions of the test setup.

(D) Schedule FAA/AUTHORITY witnessing of the test.

(E) Submit a final test report describing all results and obtain FAA/AUTHORITY approval of the report.

(iii) Definition of Environment. The SAE AE4L Committee report dated June 20, 1978, is an acceptable criteria to define the worst-case lightning strike which may be encountered by the rotorcraft in service. An additional explanation of the lightning environment may be found in FAA Report DOT/FAA/CT-89/22, “Aircraft Lightning Protection Handbook.” This handbook will assist aircraft design, manufacturing, and certification organizations in protecting aircraft against the direct and indirect effects of lightning strikes, in compliance with Federal Aviation Regulations. It presents a comprehensive test criteria to provide the essential information for the in-flight lightning protection of all types of fixed/rotary wing and powered lift aircraft of conventional, composite, and mixed construction and their electrical and fuel systems. The handbook contains chapters on the natural phenomenon of lightning, the interaction between the aircraft and the electrically charged atmosphere, the mechanism of the lightning strike, and the interaction with the airframe, wiring, and fuel system. Further chapters cover details of designing for optimum protection; the physics behind the voltages, currents, and electromagnetic fields developed by the strike; and shielding techniques and damage analysis. The handbook ends with discussion of test and analytical techniques for determining the adequacy of a given protection scheme.

On March 5, 1990, FAA Advisory Circular AC 20-136, "Protection of Aircraft Electrical/Electronic Systems Against the Indirect Effects of Lightning," was issued. Appendix 3 of that document contains an updated quantification of the severe natural lightning environment. Additionally, an AIR-100 policy letter, dated August 25, 1993, addressing multiple burst lightning strikes, should be considered. It is recommended that for new designs and applications after March 5, 1990, this revised definition of the lightning be used.

(iv) Certification Plan. The following subjects are not intended to provide a complete list of the items which should be included in the certification plan, but rather highlight some of the areas which should receive consideration. The certification plan should address the total protection which is required to allow the FADEC to continue to operate properly when the rotorcraft experiences a worst-case lightning strike encounter.

(A) Determination of Lightning Strike Attachments. Determine the locations on the rotorcraft where lightning strike attachment is likely to occur and the portions of the airframe through which currents may flow between attachments. The main and tail rotors are recognized as likely attachment points; however, consideration should be given to all possible attachment points. The swept stroke phenomenon may not exist for all lightning strike encounters due to the fact that the rotorcraft may be airborne with little or no airspeed.

(B) Establish the Lightning Environment. Establish the components of the total lightning event to be considered. These are the currents and voltages which are described in the definition of the environment.

(C) Full-Level, Complete Vehicle Testing. In accordance with traditional FAA/AUTHORITY Policy, the demonstration that the FADEC installed in a complete type design rotorcraft will continue to operate properly when exposed to a worst-case lightning strike is sufficient to demonstrate compliance with § 29.1309(a). Because of the difficulties involved in utilizing this type of an approach, it is generally not used.

(D) Analytical Processes. A description should be given in the certification plan of the analytical process and/or certification tests to be utilized to demonstrate protection effectiveness. Typically, the certification plan will include a combination of analysis and tests. (Analytical techniques are most often utilized to predict the levels of lightning-induced transients in interconnecting wiring.) In most cases, successful analyses are based upon well-defined geometrical or electrical parameters such as structural dimensions and materials resistivities. When electrical characteristics of structural materials are not well established, development tests are often utilized to obtain this data which is subsequently utilized in an analysis. In more complex structures and/or electrical/electronic system installations, it is sometimes difficult or impossible to define the problem in terms that can be analyzed. In these cases, development or verification testing is often relied upon. The purpose of the certification plan is to show how developmental tests, analyses, and verification tests are combined to demonstrate protection design adequacy. In certain cases, previously verified designs can be incorporated and their adequacy confirmed by reference to previous verifications. Such reference should also be incorporated in the certification plan.

(1) The verification testing should be conducted on a system which simulates as closely as possible the installed configuration. As few items as possible of actual hardware should be simulated.

(2) The use of various analytical processes usually requires that the system component tolerance is established. The SAE AE4L Committee Report No. AE4L-81-2 is the recommended reference to be used for the testing accomplished to determine these tolerances. The testing which is performed to determine the tolerance level of the control computer should include a consideration for the occurrence of a nonrecoverable digital upset. One method to provide this consideration is to have the unit powered and the processor operating normally under software control (usually this should be the exact software for which approval is sought) when the test is performed. If strike testing is used, then several shots should be made to develop enough data to provide a reasonable confidence level. It is an acceptable procedure for the engine manufacturer, while he is obtaining his type certificate, to accomplish this bench testing to determine the level of tolerance of the FADEC system components to lightning encounter indirect effects. This approach has the advantages that the bench tests are not necessarily required to be repeated when the engine is installed in a different airframe. This recommendation is not meant to add a requirement to the engine manufacturer but to propose a more efficient method of certification. If this tolerance was not determined by the engine manufacturer, the applicant installing the FADEC in a rotorcraft would be expected to furnish this data.

(3) For complete airframe verification testing, a minimum level of at least 1KA peak and a current rise time of 2KA/microsecond are recommended. It is often difficult to obtain valid results at lower levels due to poor signal-to-noise ratios. When complete vehicle testing is accomplished at some lower level, or through some alternate test technique such as low level swept CW testing, consideration should be given to nonlinear airframe response, diffusion effects, and alterations in current paths caused by arcing and flashover.

(4) As with any analytical method, it is prudent to include a margin of safety to account for the uncertainties involved in the analytical and testing processes. A level of 6 dB is recommended for those analyses which are confirmed by the use of reduced level, full-scale vehicle testing. This safety margin is the difference between the airframe installed system responses and the system component tolerance, not an adjustment to the quantification of the atmospheric environment. (The airframe system response to the worst-case lightning event should be at least 6 dB less than the FADEC system computer and components tolerance level. Number of dB is defined as $20\ LOG_{10}\ (V_1/V_2)$ and $20\ LOG_{10}\ (I_1/I_2)$ where V_1 and I_1 are the determined tolerance levels of the system components and V_2 and I_2 are the extrapolated airframe response.)

(5) When an analysis has no associated full-scale vehicle testing to confirm the analysis, the analysis should be very rigorous. Additionally, it should be expected in this situation that this analysis indicates a very large margin of protection. Many factors must be considered in determining what constitutes an acceptably large margin. The specific additional margin required should be based on an assessment of the inherent uncertainty of a given analysis. Approximately an additional 25 dB of protection has been deemed acceptable for a reasonably rigorous analysis performed on an airframe for which the response characteristics are known.

(E) Pass/Fail Criteria. The certification plan should address a pass/fail criteria for the testing and analysis to be performed. The following items should be satisfied to assure acceptable system performance:

(1) No immediate crew action must be required.

(2) Automatic control of the engine cannot be lost for any appreciable period of time. The engine must not be allowed to be out of control for a period of time which will result in a hazard in a worst-case flight condition. Obviously, any rapid, uncontrolled divergence is not acceptable.

(3) No crew action should be required to reset the system. This is not to imply that the system cannot be designed with a manual reset, but the manual reset cannot be used to show compliance to recover from a digital upset.

(4) The resumption of engine control after an upset must be reasonably within the range which existed before the upset.

(5) No critical data can be lost.

(6) After the system recovers, if the performance of the system has been degraded in a noncritical manner which would reduce the capability of the rotorcraft or the ability of the pilot to cope with adverse operating conditions, then the crew must be alerted to this system degradation.

(v) System Installation Considerations. In most cases, the installation of the system components is a constituent part of the lightning protection. This is particularly true in the use of shielding techniques. If these installation features are required for adequate lightning protection, consideration should be given to ensure that their effectiveness is not derogate in service. Information should be made available to the parties who service and operate the rotorcraft to allow them to take actions necessary to ensure the continued effectiveness of the system lightning protection.

(4) Lightning Protection.

(i) Background. During the original design and development of rotorcraft and the development of regulations concerning these aircraft, little attention was given to protection from the meteorological phenomenon of lightning. This was, in part, because the early aircraft were constructed mostly of metal and had little, if any, dependence on advanced technology systems. Contemporary design transport category rotorcraft are utilizing the same advanced technology systems and materials as transport category airplanes. Because of this fact, a specific requirement has been added by Amendment 29-24 for the consideration of lightning strike protection of required systems, equipment, and installations. The addition of paragraph (h) to § 29.1309 further defines the consideration required for the foreseeable operating condition of a lightning strike encounter on the rotorcraft.

(ii) Procedures.

(A) Section 29.1309(h) requires, when showing compliance with § 29.1309(a) and (b), that the effects of lightning strikes on the rotorcraft be considered.

(1) The first step in demonstrating compliance is to perform a fault/failure analysis (F/FA) to identify those functions for which the loss of function or malfunction may result in a catastrophe to the rotorcraft. An F/FA should be conducted on each system whose failure to function properly would prevent the continued safe flight and landing of the rotorcraft. These systems should be designed and installed to ensure that they can perform their intended function during and after exposure to lightning.

(2) Additionally, evaluation must be performed to identify each system whose failure to function properly would reduce the capability of the rotorcraft or the ability of the flight crew to cope with adverse operating conditions. These systems should be designed and installed to ensure that they can perform their intended function after exposure to lightning.

(3) The lightning strike models to be used for system justification should be as described in SAE AE4L Committee Report AE4L-87-3, Rev. B, dated January 1989 (or later version). The recommended reference for performing such analysis is Society of Automotive Engineers, Aerospace Recommended Practice 926A.

(B) Detailed means of compliance should be agreed with the authorities taking into account the effects on the rotorcraft and minimum considerations are as follows:

(1) Any combination of analysis and testing should be agreed with the authority.

(2) For test results, an extrapolation of the threat current parameters of more than a factor of 200 is not recommended for full scale low level pulse testing, due to the difficulty of obtaining valid results with poor signal-to-noise ratios.

(3) For a proven analysis technique, a safety factor of at least 2 will be necessary.

(C) Flight and engine controls are examples of “critical” functions and with these critical functions defined, an analysis and/or testing should be performed to show compliance with § 29.1309(a); i.e., equipment, systems, and installations performing those identified functions should be designed and installed to ensure that they continue to perform their intended functions considering the conditions of the rotorcraft experiencing a worst-case lightning strike encounter. Section 29.610 contains some methods which may be utilized for less complex mechanical systems; however, a great deal of difficulty will be experienced in trying to use these criteria to demonstrate that a very complex avionic system complies with § 29.1309(a). These avionic systems thus identified usually only require protection from indirect effects of lightning. If it is determined such is the case, then a method as outlined in paragraph AC 29.1309b(3), Lightning Strike Protection of Full Authority Digital Engine Controls, is recommended. This method may be readily adapted to other avionic systems performing critical functions. Also, this identifies an acceptable quantification of the expected airborne environment. The next step involves expanding the F/FA to determine if the malfunctioning of several “essential” systems in relation to other systems would result in a hazard to a Category B rotorcraft or preclude the continued safe flight and landing of a Category A rotorcraft. If groups of functions are so identified, sufficient lightning protection should be provided to prevent a hazardous malfunction situation on Category B rotorcraft or provide those conditions which prevent continued safe flight and landing on a Category A rotorcraft are extremely improbable. In performing this part of the analysis, attention should be given to the fact that many of the required equipment, systems, and installations may fail simultaneously with other required equipment, systems, and installations and result in a reduction of the capability of the rotorcraft but still not result in a catastrophe. An example of required equipment for which the simultaneous failure of all the required equipment is catastrophic is a failure which results in a total loss of attitude display for IFR certified rotorcraft operating in instrument meteorological conditions. Note that the analysis which is utilized to demonstrate that these failures are extremely improbable should have the encounter with a worst-case lightning strike as a given event; i.e., probability is unity. Additionally, for a Category A rotorcraft, an autorotation is not considered continued safe flight and landing.

c. Failure Analyses.

(1) Power and distribution systems should be analyzed to show compliance with § 29.1309.

(i) One acceptable procedure for documenting the analysis is contained in Society of Automotive Engineers (SAE) Fault/Failure Analysis Procedure ARP 926A, revised November 15, 1979.

(ii) As a minimum, any analysis should consider the effect of failures of components and systems on the capability of the rotorcraft to perform its intended function without hazard.

(iii) The analysis should consider the indication of failure. Those latent failures that occur without indication should be considered in all possible sequences and combinations of additional failures until a positive indication of failure is provided.

(iv) The analysis should consider failure of indirectly related parts of installations which could induce failure in the system being analyzed. For example: the effect of hydraulic fluid sprayed on electrical components as a result of a ruptured hydraulic line. Another example is the result of a ruptured bleed air line and its effect on hydraulic, fuel, or electrical lines/cables.

(v) The Type Inspection Authorization (TIA) should call for specific simulated failures, evaluation of failure detection, failure warning, and performance of the remaining system on the ground and in-flight to verify the critical aspects of the failure analysis. The applicant should provide a proposed detailed test procedure for incorporation in the TIA to accomplish this verification. The applicant's proposed tests simulating in-flight failures should be carefully reviewed by both the systems engineer and flight test pilot to assure the flight test crew will not be subjected to hazardous flight. Where practicable those simulated failures that would be hazardous in flight should be evaluated by ground tests. Analyzed and tested systems (where functioning is required) exhibiting hazards or failing to perform their intended functions under any foreseeable operating conditions must be redesigned to comply with § 29.1309.

(2) Utilization systems that are required or critical as to performance of intended function or result in rotorcraft hazard upon failure should also be analyzed for failures by the procedures of paragraphs c(1)(a) through c(1)(d) above. Examples of systems which may be critical are autopilots, hydraulic control systems, navigation and flight instruments on IFR approved rotorcraft, and bleed air systems.

d. Safety Assessment. Caveat: The safety assessment process that contains the Functional Hazard Assessment addressed herein is a methodology of general application to show compliance for systems, and all or parts of it may be applied as necessary depending on the complexity of design. The concept of safety assessment is provided by SAE document ARP 4754, "Systems Integration Requirements" and this aspect of this document is endorsed as an acceptable way of showing compliance to the requirements for function/design assurance. SAE document 4761, "Safety Assessment Guidelines for Civil Airborne Systems and Equipment" can be used as a guide to perform the different analysis that forms the safety assessment. The starting point is a Functional Hazard Assessment (FHA) that identifies the critical functions and in turn the systems or parts of systems that provide these critical functions. Further analysis is directed to be performed on the identified sources of these functions (i.e., system(s) performing these function(s)). One of the most common directed analysis is the reliability analysis (Failure Mode and Effects Analysis) and it is a bottom-up analysis. The safety assessment process consists of several different types of analysis; only two are addressed by this AC at this time. The other analysis will be addressed in a future revision of this AC.

(1) Functional Hazard Assessment. The FHA is a top down assessment that, on an aircraft level, starts at flight operations and identifies the criticality levels associated with the malfunction/ failure of functions that can have safety effects. The source of these functions is also identified by the FHA. There are five criticality levels defined in subparagraph f, "Computer Software" of this paragraph as they pertain to FAR Part 27 and 29. These level definitions are used in the FHA to decide which level matches the malfunctions/failures under consideration. The bottom-up analysis is then directed to the identified function source. Reliability analysis is the most common of the bottom-up analysis.

(2) Reliability Analyses. Numerical Reliability Analyses may be developed, on an optional basis, as a continuation of the failure analysis procedure.

(i) Specific reliability numbers are not shown in § 29.1309. The necessary degree of reliability is a function of the criticality of the system under consideration. Acceptable sources of component failure rates are (1) military service records or handbooks, such as MIL-HDBK-217C, (2) operator or manufacturer service records, such as airline records on sufficiently similar component designs, and (3) laboratory life tests.

(ii) For the purpose of conducting or evaluating an analysis, the following terms and numerical values should apply:

(A) FLIGHT TIME (Block Time). The time from the moment the rotorcraft first moves under its own power for the purpose of flight until the moment it comes to rest at the next point of landing.

(B) PROBABILITY CLASSIFICATIONS. Five probability classifications are defined below, although the rules refer to three. This is not considered a problem as long as the conversion below is applied, and it is recognized that only the divisions within the same range are different. Quantitative ranges are also provided as a common point of reference if numerical probabilities are used. The quantitative ranges given for these classifications are considered to overlap due to the inexact nature of probability estimates. When assessing the acceptability of a failure condition using a quantitative analysis, the numerical ranges given below should normally be interpreted to be the allowable risk for an hour of flight time based on a flight of mean duration for the rotorcraft type. However, when assessing a function which is used only at a specific time during a flight, the probability of the failure condition should be calculated for the specific time period and expressed as the risk for the flight condition, takeoff, landing, etc., as appropriate.

(1) FREQUENT. (This is the lower part of the range 10^{-5} or greater previously applied to the term "PROBABLE".) Frequent events may be expected to occur several times during the operational life of each rotorcraft, that is based on a probability on the order of 10^{-3} or greater.

(2) REASONABLY PROBABLE. (This is the upper part of the range 10^{-5} or greater previously applied to the term "PROBABLE".) Probable events may be expected to occur several times during the operational life of each rotorcraft, that is based on a probability on the order of between 10^{-3} to 10^{-5}.

(3) REMOTE. (The term "REMOTE" is not related to the structural use of the term.) (This is the lower part of the range 10^{-9} to 10^{-5} previously applied to the term "IMPROBABLE".) Remote events are not expected to occur during the total operational life of a random single rotorcraft of a particular type, but may occur during the total operational life of all rotorcraft of a particular type, that is based on a probability on the order of between 10^{-5} to 10^{-7}.

(4) EXTREMELY REMOTE. (The term "REMOTE" is not related to the structural use of the term.) (This is the upper part of the range 10^{-9} to 10^{-5} previously applied to the term "IMPROBABLE".) Extremely remote events are not expected to occur during the total operational life of a random single rotorcraft of a particular type, but may occur during the total operational life of all rotorcraft of a particular type, that is based on a probability on the order of between 10^{-7} to 10^{-9}.

(5) EXTREMELY IMPROBABLE. (Remains the same for both the three or the five classifications.) Extremely improbable events are so unlikely that they need not be considered to ever occur, unless engineering judgment would require their consideration. A probability on the order of 10^{-9} or less is assigned to this classification.

NOTE:

If a quantitative analysis is used to help show compliance with Regulations for equipment which is installed and required only for a specific operating condition for which the rotorcraft is thereby approved, credit may not be taken for the fact that the operating condition does not always exist. Except for this limitation, appropriate statistical randomness of environmental or operational conditions may be considered in the analysis. (However, the particular condition and probability of that condition should be agreed to with the FAA/AUTHORITY.)

The five probability terms defined in paragraph d(2)(ii)(B) above are intended to relate to an identified failure condition resulting from or contributed to by the improper operation or loss of a function or functions. These terms do not define the reliability of specific components or systems.

Generally, the guidance for Failure Analysis of paragraph c, is not required in its entirety for Category B, non IFR rated rotorcraft. The only failure/reliability requirement is that no single failure can result in a hazard to the rotorcraft. This can usually be accomplished by a systems safety assessment that may or may not, depending on complexity and configuration, require a numerical reliability analysis.

e. Documentation. All laboratory, ground and flight tests, and failure analyses, must be documented in sufficient detail to show compliance with § 29.1309 and included in the type design file. Section 21.31(a) provides the regulatory basis for requiring this documentation. If the applicant elects to use a numerical reliability/probability analysis it must also be documented in sufficient detail.

f. Computer Software. The latest standard for qualification of software is DO178B; however, the use of DO178A for a standard is not precluded at this time. Because of this dual standard situation, at this time, use of both standards will be addressed for qualification of software that is used for airborne systems and equipment certification.

(1) RTCA Document DO178A

(i) RTCA Document DO-178A, "Software Considerations in Airborne Systems and Equipment Certification," dated March 22, 1985, is a recommended standard to be used for the approval of system software. This document defines three levels of software; i.e., levels 1, 2, and 3. The level of the software is related to the consequence of a system malfunction caused by an error in the software. The criticality categories are:

(A) Critical - Functions for which the occurrence of any failure condition or design error would prevent the continued safe flight and landing of the aircraft.

(B) Essential - Functions for which the occurrence of any failure condition or design error would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions.

(C) Nonessential - Functions for which failures or design errors could not significantly degrade aircraft capability or crew operational cue.

(ii) The different software levels may be related to the criticality categories. Level 1 software, the most error free software, is usually required for critical functions. However, Level 1 software may sometimes be reduced by system architecture techniques such as the use of redundant (dissimilar) software performing the same function. Level 2 software is required for essential functions. It should be noted that those systems, equipment, and installations, with functioning required by 14 CFR subchapter C, are by this definition, essential functions. The criticality of the function should be determined by the use of a fault/failure or hazard analysis. The

Society of Automotive Engineers Aeronautical Recommended Practice Document Nos. 926A and 1834, are the recommended references for performing these analyses.

CAVEAT: The user of DO-178A is cautioned by a caveat in Chapter 3 that for a certain class of systems, the techniques in DO-178A, Level 1 software are not by themselves sufficient consideration for reliance on system software to preclude a catastrophic event. Additional considerations are required with this class of system for software verification and validation (V&V) in addition to those required for DO-178A Level 1. This class of systems is one which has been called, “full flight regime critical.” An example of such a system is a fly-by-wire flight control. This system must perform its intended function through the full flight regime to provide for the continued safe flight and landing of the rotorcraft. For this system, software and system level validation beyond the scope of DO-178A are required. Also, DO-178A cautions the user against the assignment of probabilities of residual software errors. The conclusion of Special Committee No. 152 (the RTCA committee that wrote DO-178A) was that the present methods available for assigning “reliability” numbers to software do not yield credible results for certification purposes.

(2) RTCA Document DO-178B

(i) RTCA Document DO-178B, “Software Considerations in Airborne Systems and Equipment Certification,” dated December 1, 1992, is the latest standard and is recommended to be used for qualification and subsequent approval of system software. This document defines five levels of software; i.e., levels A, B, C, D, and E. The level of software is related to the criticality of the function that may be adversely affected by an error in the software. The criticality categories are as follows:

(A) Catastrophic - failure conditions that would prevent continued safe flight and landing.

(B) Hazardous/Severe-Major - failure conditions that would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions to the extent that there would be:

(1) A large reduction in safety margins or functional capabilities.

(2) Physical distress or higher workload such that the flight crew could not be relied on to perform their tasks accurately or completely.

(3) Adverse effects on occupants, including serious or potentially fatal injuries, to a small number of those occupants.

(C) Major - failure conditions that would reduce the capability of the aircraft or the ability of the crew to cope with adverse operating conditions to the extent that there would be, for example: a significant reduction in safety margins or functional capabilities, a significant increase in crew workload or in conditions impairing crew efficiency, or discomfort to occupants, possibly including injuries.

(D) Minor - failure conditions that would not significantly reduce aircraft safety, and would involve crew actions that are well within their capabilities. Minor failure conditions may include, for example: a slight reduction in safety margins or functional capabilities, a slight increase in crew workload, such as, routine flight plan changes, or some inconvenience to occupants.

(E) No Effect - failure conditions that do not affect the operational capability of the aircraft or increase crew workload.

(ii) The software levels are usually related to the criticality categories. Level A qualified software is the most error-free software and is usually required for functions that could exhibit catastrophic failures. However, additional considerations may moderate this direct relationship and allow some lower level of software qualification for higher function criticality categories. Some of the moderating factors may be architecture of the system/software, redundancy of systems using dissimilar software, hardware/software monitors, and independent function contributions. It is recommended that these practices are carefully employed and prior FAA/AUTHORITY approval of the methodology should be obtained before the design is pursued. Typically, Level A is required for flight controls where the catastrophic criteria applies. Level B qualification is less than Level A and is employed in some flight controls and required flight instruments, where their malfunction would result in the hazardous/severe major critical category effect on the rotorcraft or its crew and occupants. Level C qualification is less than Level B and is employed in some required flight instruments commensurate with failure category “Major.” However, this lower level of qualification may be appropriate for either or both of the higher failure categories if the criteria can be met as discussed previously for level reduction. Level D qualification is less than Level C and is typically employed for required systems/equipment that do not exhibit the fault potential of the higher categories. Level E qualification is less than Level D and is employed in those systems/equipment that are not required by the regulations. Examples are entertainment systems, powered seats, etc.

(iii) Although a rough correlation exists between software levels and criticality levels that in turn relate to a probability in numbers, these numbers cannot be applied to software to determine reliability. The quality assurance of software is obtained by the processes delineated in DO178B and at this time, have no correlation with probability.

g. High Intensity Radiated Fields (HIRF).

(1) Explanation. A regulatory project is active to add requirements for the protection of aircraft electrical and electronic systems from the effects of the HIRF environment. This effort is the result of technological advances in airframe and

electronic systems design and a concurrent increase in the levels of radiated power in the aircraft environment. These changes have raised vulnerability to the electromagnetic environment of the electrical and electronic systems which perform critical and essential functions. In current type certification programs involving advanced electrical and electronic systems the FAA/AUTHORITY has adopted special conditions to provide an adequate level of safety.

(i) The special conditions are directed toward the operation and operational capability of the installed electrical and electronic systems that perform critical functions. The applicant may demonstrate that these systems are not adversely affected when the aircraft is exposed to the HIRF environment, or as an alternative a laboratory test may be conducted, as discussed in the "Discussion" associated with each special condition. The laboratory tests would be conducted at a peak electromagnetic strength of 100 or 200 volts per meter, as appropriate, in a frequency range of 10 kHz to 18 GHz.

(ii) A definition of the HIRF environment is included in FAA Notice 8110.71, dated 2 April 1998 [Subject: Guidance for the Certification of Aircraft operating in High Intensity Radiated Field (HIRF) Environments]. SAE chartered a HIRF committee, SAE-4R, to define the HIRF environment for VFR and IFR aircraft, to prepare a Users Manual, and to submit a proposed advisory circular to the FAA/AUTHORITY. This committee has currently completed its task, except for the Users Manual. The referenced FAA Notice has been updated several times to reflect the latest SAE-4R recommended electromagnetic fields.

(iii) If the laboratory test alternative is selected the 100 volts/meter level is considered appropriate for a function that is critical during IFR operations and the 200 volts/meter level is considered appropriate for a function that is critical during VFR operations. This is because the minimum en route altitude for IFR flight is 1,000 feet or 500 feet (FAA or ICAO), and rotorcraft operating VFR can and do operate regularly at lower altitudes. The attitude system is an example of a system performing a critical function during IFR operation. A full authority digital engine control (FADEC) system is an example of a critical function during VFR and IFR operation.

(iv) JAA has issued a HIRF Interim Policy Paper to be used as JAA special conditions. This Interim Policy Paper was drafted by the Aviation Authority members of EUROCAE WG 33. The JAA Interim Policy requires full aircraft testing for systems performing Level A control functions and laboratory testing for Level A display, Level B, and Level C systems. The Interim JAA HIRF environment is the January 1992 environment used in INT/POL/25/2, with exceptions in the 2-4GHz and 6-8GHz bands. JAA guidance material for acceptable means of compliance is AMJ.1317.

(2) <u>Procedures</u>. It is recommended that the applicant present a plan to the cognizant FAA Aircraft Certification Office (ACO) for approval, outlining how the compliance with the HIRF requirements will be attained. This plan should also propose a pass/fail criteria for the operation of critical systems in the HIRF environment.

(i) A preliminary hazard analysis should be performed by the applicant for approval by the cognizant FAA ACO to identify electrical and/or electronic systems that perform critical functions. The term “critical functions” means those whose failure would prevent the continued safe flight and landing of the rotorcraft.

(ii) The systems performing critical functions that are identified by the preliminary hazard analysis are candidates for the application of HIRF requirements. A system may perform both critical and non-critical functions; however, the HIRF requirements only apply to critical functions. If redundant systems are used, all systems should be subjected to test/analysis for the HIRF requirements.

(iii) The latest revision of RTCA document DO-160/EUROCAE ED-12D, Section 20, is an appropriate reference for laboratory test procedures. In addition, a separate advisory circular and users guide on the subject of HIRF is being drafted for the FAA/AUTHORITY by the SAE AE4R Subcommittee.

SUBPART F - EQUIPMENT

INSTRUMENTS: INSTALLATION

AC 29.1321. § 29.1321 (Amendment 29-21) ARRANGEMENT AND VISIBILITY.

a. Background. This section is the first in a series that concerns the installation of instruments. Specific requirements for individual instruments are addressed in other paragraphs. The instruments should be arranged in a manner such that the pilot may avail himself of the information displayed by the instruments without undue distraction. Additionally, for instrument flight, the rule requires that the attitude, altitude, airspeed, and compass indicators be grouped in the so-called standard "T."

b. Procedures.

(1) For rotorcraft certified for VFR operation, the flight, navigation, and powerplant instruments should be placed such that the pilot and copilot, if a required crewmember, can easily see and read these instruments when seated normally. Additionally, the instruments should be located so that the necessity for the pilot to turn his head is minimized. The instruments which are necessary for safe operation including the airspeed indicator, gyroscopic direction indicator, gyroscopic bank-and-pitch indicator, slip-skid indicator, altimeter, rate-of-climb indicator, rotor tachometers, and the indicator most representative of engine power should be installed immediately in front of the pilot.

(2) The other powerplant instruments should be grouped together and visible to any appropriate crewmember. On multiengine rotorcraft, there should be no confusion regarding which engine an individual gauge represents. This is usually accomplished by mounting the engine gauges vertically in the center of the instrument panel. Identical gauges are placed next to each other and positioned from left to right in the same position and sequence as the corresponding engine location in the airframe.

(3) An evaluation should determine that vibration of the instrument panel does not exceed the tolerances of the instruments. The instrument manufacturer will usually provide data that indicate the level of vibration for which the instrument has been qualified. The flight test evaluation of the rotorcraft should explore and determine that the vibration of the instrument panel does not affect the readability of the instruments. To meet these two criteria, it has been necessary in some installations to "shock mount" or otherwise isolate the instrument panel.

(4) The flight test evaluation should also determine that the flags or malfunction indicators of the instruments should be readily visible in all combinations of lighting for approved kinds of operations.

(5) For IFR-certified rotorcraft, there is an additional requirement that the airspeed, altimeter, attitude, and compass instrument be located in a standard “T” configuration in front of the pilot. This configuration is:

airspeed - attitude - altitude

compass

(AC 29 Appendix B further addresses IFR panel arrangement.)

(6) Geometric variation from a perfect “T” has been permitted. Each installation should be evaluated for suitability based on criteria such as panel size, ease of scan, and readability of the individual elements in the overall presentation. Advisory Circular 25-11, Transport Category Airplane Electronic Display Systems, provides additional guidance for “glass cockpit” installation.

AC 29.1322. § 29.1322 (Amendment 29-12) WARNING, CAUTION, AND ADVISORY LIGHTS.

a. Explanation.

(1) Cockpit devices are color coded to symbolically represent various functions and varying levels of importance for flight crew operation. From early times, an attempt has been made to take full advantage of associations developed early in life as a result of continuous exposure to our daily environment.

(2) Military design specifications were the first to reference color-coding in cockpit design requirements. In the mid-1940s, the CAA initiated the first color-coding requirements for civil cockpit design. Color-coding standards for cockpit visual signals soon followed. MIL-STD-411, May 31, 1957, identified three separate categories of light signals:

(i) Warning Light - indicates the existence of a hazardous condition which may require immediate corrective action.

(ii) Caution Light - serves to alert the operator to an impending dangerous condition requiring attention but not essential equipment, or attracts attention for routine purposes.

(iii) Advisory Light - indicates safe or normal configuration, condition of performance, or operation of essential equipment, or attracts attention for routine purposes.

(3) Examples of warning and caution signals were included in later versions of the military standard, and a few of those are shown below:

Warning Signals	Caution Signals
Cabin Pressure Failure	Trim Failure
Fire	Fuel Low
Fuel System Failure	Generator Inoperative
Landing Gear Unsafe	Defrosting Failure

(4) Specific color designation for civil advisory lights was first addressed in Amendment 3 to the Rotorcraft Certification Rules (Parts 27 and 29) on January 19, 1968, with adoption of new §§ 27.1322 and 29.1322.

(5) In subsequent revision (Amendment 29-12), green lights were redesignated and additional colors introduced for flexibility in the requirement.

(6) Green signifies a safe operating condition and more specifically has come to signify landing gear extended and locked. Extensive use of green annunciators throughout the cockpit should generally be avoided due to possible confusion with the special use of green for landing gear. If green annunciators are physically and functionally removed from the landing gear operation, they may be found acceptable for a variety of "safe operating" applications. One such application is "all green for approach," used in autopilot, flight director, and other navigation system displays.

(7) Other colors may be utilized as advisory lights in accordance with § 29.1322(d). Red and amber must not be used as advisory lights due to the possibility of introducing confusion into the cockpit. Obviously, yellow and pink annunciators should be avoided due to their similarity to amber and red. White and blue have been successfully utilized as advisory segments in past civil designs.

(8) The primary test for designation of color is:

(i) Red - Is immediate action required:

(ii) Amber - Is pilot action (other than immediate) required?

(iii) Green - Is safe operation indicated, and is the indication sufficiently distinct to prevent confusion with the landing gear down indication?

(iv) Other advisory lights - Is the meaning clear and distinct enough to prevent confusion with other annunciators? Do the colors which are utilized differ sufficiently from the colors specified in paragraphs (b)(1), (2), and (3) above?

(9) Annunciator lights should be visible during bright daylight conditions. This should include visibility in direct sunlight unless lights are located in such a manner that direct sunlight cannot impinge on them.

(10) If dimming capability is provided, all annunciators, including master warning and caution, may be dimmable so long as the annunciation is clearly discernible for night operation at the lower lighting level. Undimmed annunciations have been found unacceptable for night operation due to disruption of cockpit vision at the high intensity. The dimming circuit should automatically revert to the high intensity setting when power is removed. Automatic dimming/brightening through the use of a photo cell is also acceptable, as are circuits which enable a dimming switch through a position light or other cockpit lighting controls.

(11) The use of flashing lights should be minimized. If a flashing feature is used, it should be controllable through pilot action so that flashing annunciation does not persist indefinitely. The indicator should be so designed that if it is energized and the flasher device fails, the light will illuminate and burn steadily.

(12) The activation of caution and warning lights should readily attract the attention of the appropriate crewmember while performing duties under both normal and high workload conditions.

b. <u>Procedures</u>.

(1) Red shall be reserved for annunciation of emergency conditions requiring immediate corrective action. Typical examples include fire, transmission oil pressure, engine failure, and battery overheat. The use of red for annunciators which do not require immediate action must be avoided. Use of red when it is not needed tends to lessen the impact of a red annunciator and the needed pilot association for immediate action. In evaluating cockpit annunciators for acceptability, the FAA/AUTHORITY should assure <u>all</u> annunciators which require immediate action are red and that <u>only</u> those requiring such action are red. If a master warning light is provided, it should be red, and should be powered by the same signal that powers any of the individual red warning signals. An aural warning may accompany visual warning signals to enhance pilot response. Care should be taken that any aural signal is sufficiently distinct from other aural warnings, such as low rotor RPM, to prevent confusion and to assure proper crew response. A means to deactivate and reset the master warning (visual and aural) is required. Resetting the master warning must not deactivate any individual warning signal.

(2) Amber shall be reserved for indicating malfunction or failure conditions which do not require immediate crew action to assure safe flight. Typical examples include door unlatched, inverter failure, generator failure, fuel filter clogged, and parking brake engaged. Amber should generally be utilized for malfunction and failure conditions which do not <u>require</u> immediate action. The key word here is “require.” Obviously, a pilot should perform corrective action for malfunction or failure conditions in a timely manner as soon as other cockpit priorities allow. The time increment associated with “immediate action” may vary with the system involved, the flight regime, and the aircraft; however, 15 seconds is a representative value in evaluating this term. This by no means indicates that any red annunciator can be ignored for 15 seconds.

For red annunciators, some type of immediate pilot response is expected. If immediate pilot action is not required, the FAA/AUTHORITY should recommend the use of an amber designation. If a master caution light is provided in addition to a master warning light, the master caution annunciator should be amber, and should be powered by the same signal that powers any of the individual amber caution signals. Reset considerations for the master caution are the same as those detailed above for the master warning.

c. Annunciator Panel Design

(1) Explanation.

(i) The annunciator panel design should be reviewed for the presence of failure modes that can cause illumination of multiple panel segments.

(ii) Many test circuits that are diode isolated are vulnerable to this condition. A typical sequence begins with the shorting of a test circuit diode. This failure is undetectable and goes unnoticed until an actual failure condition occurs which causes the associated panel segment to illuminate. At this time all panel segments connected to the test circuit will illuminate.

(iii) This configuration becomes a special problem when one or more of the panel segments are red. A red light calls for immediate action by the crew, and the crew does not have adequate information for immediate action when many false panel segments are illuminated.

(iv) If the design review indicates a problem, a redesign of the panel to eliminate the condition is considered to be the best solution and is highly encouraged.

(2) Procedures.

(i) An alternative to panel redesign might be the following:

(A) Review the annunciator panel design and note which segments are red.

(B) Determine if cross reference information is available in the cockpit to allow elimination from consideration of any the red segments. (Example: Red low fuel pressure light and low fuel pressure gauge. Normal operation of the gauge would be a reason to assume the light did not cause the problem.)

(C) Where a cross reference is available, further design review of that function is not necessary; however, it may be appropriate to include procedural information in the emergency procedures section of the rotorcraft flight manual.

(D) If cross references are not available for red segments, additional isolation should be incorporated into the annunciator design for those functions.

(ii) If cross referencing is not practical the following approach is encouraged.

(A) Review the annunciator panel design and note which segments are red.

(B) Determine if isolation diodes are checked during the application of battery or external power before starting the engines. (Example: Red low oil pressure light. If isolation diode is shorted, all panel segments will light as soon as battery or external power is applied.)

(C) When the isolation diode can be checked before starting engines, further design review is not necessary.

(D) If diodes are not automatically checked before starting, then additional isolation, should be considered.

(3) Annunciator Panel Arrangement. The annunciator panels should be arranged in a logical manner to reduce the crew's time required to locate faults and to increase their efficiency in following Aircraft Flight Manual procedures. For example, engine annunciators on multiengine rotorcraft should be physically located on the panel to coincide with engine location (left or right) so that properly operating engines are not inadvertently shut down due to crew confusion over which engine has malfunctioned.

AC 29.1323. § 29.1323 (Amendment 29-3) AIRSPEED INDICATING SYSTEM.

a. Explanation.

(1) The accuracy of all flight test data concerned with the velocity of the rotorcraft is dependent on the calibration of the airspeed indicating system. For this reason, the airspeed system position error should be determined very early in the program.

(2) Since air density varies with altitude, the speed reading will only be correct under standard sea level conditions. However, in an actual installation, the indicator reading, even under standard sea level conditions, may differ from the calibrated airspeed because the static system does not sense true static pressure. This error in detection of static pressure is called position error. It is caused by the pressure field built up around the rotorcraft in flight. This pressure field will vary in intensity with dynamic pressure making the position error a function of calibrated airspeed. Since airspeed information is presented to the crew in terms of indicated airspeed, it is necessary to determine the position error for the rotorcraft to be flown safely.

b. Procedures.

(1) There are different methods to determine position error such as trailing bomb, airspeed course, boom system, and so forth. Each method has its own advantages and disadvantages, but will yield satisfactory results if done correctly. The airspeed system should be calibrated throughout the airspeed range of the rotorcraft and under the various flight conditions of cruise, climb, and autorotation standard. In addition, the effects of gross weight and center of gravity should be investigated.

(2) It may also be necessary to recalibrate the system with a change in external configuration if such a change may affect the airflow near the pitot or static sources.

(3) Additional information regarding position error is included in AC 29 Appendix B b(10) and should be considered if pursuing an IFR approval.

(4) Static system installation information is included in paragraph AC 29.1325. Technical Standard Order (TSO) C16, Airspeed Tubes (Heated), gives minimum performance standards for pitot tubes, and pitot tubes qualified to this TSO normally allow for a satisfactory aircraft installation.

(5) The calibration requirements of the standard seem to be self-explanatory and are not discussed further in this paragraph.

AC 29.1323A. § 29.1323 (Amendment 29–24) AIRSPEED INDICATING SYSTEM.

a. Explanation. Amendment 29-24 to the regulations provides the requirements for Category A and Category B and defines the maximum allowable error for both.

b. Procedures. All of the policy material pertaining to this section remains in effect. In addition, calibration should be determined in level flight speeds of 20 knots and greater, and over an appropriate range of speeds for flight conditions of climb and autorotation; and takeoff. The takeoff calibration should be repeatable with respect to field lengths defined in the flight manual and avoidance of height-speed limiting envelope defined in § 29.79. Calibration errors, excluding instrument errors, may not exceed the following:

(1) Category A - Three percent or 5 knots, whichever is greater, in level flight at speeds above 80 percent of takeoff safety speed; and 10 knots in climb at speeds from 10 knots below takeoff safety speed to 10 knots above V_Y.

(2) Category B - Three percent or 5 knots, whichever is greater, in level flight at speeds above 80 percent of the climb-out speed attained at 50 feet when complying with § 29.63.

AC 29.1325. § 29.1325 (Amendment 29-24) STATIC PRESSURE SYSTEMS.

a. Explanation.

(1) This section, in conjunction with § 29.1323, provides minimum performance standards for static pressure systems. The standard provides some relief when considering the icing environmental condition in that it allows the use of an alternate static port to account for the icing condition.

(2) The standard for the consideration of environmental conditions is § 29.1309(a).

(3) The standard for consideration of malfunction conditions is § 29.1309(b).

(4) For rotorcraft that will be approved for IFR operation, the provisions of Appendix B VIII(b)(5) of Part 29 as discussed in paragraph AC 29 Appendix B, should also be considered.

b. Procedures. The installation of the static system should consider the following:

(1) Static lines should be initially routed upward immediately behind the static pressure port. This procedure will minimize the entry of moisture into the system when operating in rain or washing the rotorcraft.

(2) Drain(s) should be located at low points in the system. Line routing and clamping should allow for all moisture that does enter the system to be routed to the drain(s).

(3) If independent systems are provided, the placement of each system component should allow for maximum practicable separation of each system. As much as possible, one system should be on one side of the rotorcraft and the second system on the opposite side.

(4) Most static pressure ports that are provided for IFR operation are heated. Before any tests are conducted, a program to qualify the heater on the port should normally be agreed upon through discussions between the FAA/AUTHORITY and the applicant. It is suggested that the requirements of TSO C16, Airspeed Tubes (Heated), be used as a guide for these discussions. If the ports are not to be heated, a comprehensive analysis should be prepared, and limited testing should be conducted to verify the analysis.

(5) Other static system considerations are included in paragraphs AC 29.1323 and AC 29 Appendix B.

AC 29.1327. <u>§ 29.1327 MAGNETIC DIRECTION INDICATOR</u>.

a. <u>Background</u>. This section contains specific requirements regarding installation and functioning of a magnetic direction indicator. The magnetic direction indicator (commonly referred to as a compass) described by this paragraph is the unit required by § 29.1303(c) or the unit or system required for IFR operation by Appendix B VIII(a) to Part 29. Both of these indicators provide the pilot with an aircraft heading which is referenced to the earth's magnetic field. The unit required by § 29.1303(c) is the indicator commonly referred to as a "whiskey compass." This unit was given this designation because early units were constructed using alcohol as the medium in which the compass ball floats. This unit is generally approved as meeting the requirements of TSO-C7c. The indicator required by Appendix B to Part 29 is usually a system of units which meets the requirements of TSO-C6c.

b. <u>Procedures</u>. In showing compliance to § 29.1327(a), generally the magnetic indicator and its respective components will be tested to an appropriate standard such as RTCA DO 160B for use in a rotorcraft. If the unit functioned properly as described in the TSO during this testing, then no additional evaluation is generally required concerning vibration immunity. To determine the immunity of the indicator (system) from magnetic effects and its installed accuracy, a ground and flight test should be performed. This test should turn the rotorcraft a full 360° heading change in 45° increments. The indicator should not have an error in excess of 10° on any of the 45° increments. When performing these tests, the electrical equipment and systems should be functioning normally, and the effect of windshield heating (if installed) should be investigated. The results of the investigation may be used to construct the calibration placard which is required by § 29.1547. It should be noted that a calibration placard has not been traditionally required for slaved compass systems. Also, it should be emphasized that other aspects of the functioning and installation of these indicators should comply with the other general requirements (i.e., §§ 29.1301, 29.1309, 29.1555, etc.).

AC 29.1329. <u>§ 29.1329 (Amendment 29-24) AUTOMATIC PILOT SYSTEMS</u>.

a. <u>Explanation</u>. The automatic flight control systems used on most modern rotorcraft often perform two different and distinct functions when viewed from a regulatory compliance aspect. These two functions are augmentation of the stability of the rotorcraft and a pilot aid in maintaining attitude, altitude, and airspeed, or in radio navigation tasks. The first function of stability is not covered in § 29.1329 but is included under § 29.672. The second function as a pilot aid is the automatic pilot function covered by this section. The following procedure discusses only those parts or systems which are installed as a pilot aid. Paragraph AC 29 MG 3 discusses the use of automatic systems for Category II approaches, and paragraph AC 29 Appendix B discusses the evaluation of stability augmentation systems.

b. <u>Procedures</u>.

(1) General.

(i) The automatic pilot system should be evaluated to demonstrate that it can perform its intended function of flying the rotorcraft and that it complies with the installation, operation, and malfunction requirements of § 29.1329. In demonstrating malfunctions of the autopilot system, generally servo actuator hardovers are the most critical malfunction. If this is the case and the autopilot system utilizes the same servos and servo amplifiers as the stability augmentation system (SAS) and the autopilot function cannot produce a more severe hardover than the SAS, then no additional consideration is required for this malfunction. An evaluation using the guidance in paragraph AC Appendix B would be sufficient.

(ii) There have been autopilots approved that require the use of a monitor since they cannot meet the hardover malfunction requirements. These approvals have involved a finding of equivalent safety that is beyond the scope of this AC. Such findings of equivalent safety are made on a case-by-case basis. If an applicant is considering such a design, the applicant and the approving office should contact the Rotorcraft Standards Staff specialists for guidance.

(iii) The rule specifies that unless there is automatic synchronization, there should be some method to indicate the alignment of the actuating device to the pilot. The intent of this requirement is to provide a means such that the pilot does not inadvertently engage the system into a hardover condition. One method of achieving this has been the use of servo force meters. These meters monitor the current into the servo motor and indicate to the pilot if a signal is being sent to the servo prior to system engagement.

(iv) Various autopilot systems have used a preflight test to ensure adequate reliability. The question which often arises is: Should the preflight test function be interlocked so the autopilot cannot be engaged if the preflight test has not been accomplished? The guidance used in the past to answer this question is: If the preflight test is simple and rapid enough that the pilot may reasonably be expected to perform such a test, then it is not required to be interlocked. If, however, the preflight test is very complicated and lengthy and a pilot who was pressed by a schedule might skip such a test, then this preflight test should be interlocked.

(v) Most of the autopilots that have been approved utilize series actuators or servos such as those required for a SAS. However, this does not preclude the approval of an autopilot that uses outer loop parallel actuation. This type of autopilot may be particularly helpful in a VFR aircraft.

(2) Cockpit controls. Evaluation of the cockpit controls should include the following items:

(i) Location of the automatic pilot system controls are such that their operation is properly labeled and is readily accessible to the pilot(s).

(ii) Annunciator colors conform to the colors specified in § 29.1322 (reference paragraph AC 29.1322).

(iii) A determination is made that the controls, control labels, and placards are readable and discernible under all expected cockpit lighting conditions.

(iv) Motion and effect of the autopilot cockpit controls should conform with the requirements of § 29.779.

(v) Any disconnect of the autopilot should be annunciated.

c. Malfunction Evaluations. To preclude hazardous conditions that may result from any failure or malfunctioning of the autopilot the following failures should be evaluated. This evaluation should also account for any hazards that also might be caused by inadvertent pilot action. The guidance in paragraph AC 29 Appendix B should be used to determine the appropriate reaction times of the human pilot to an autopilot malfunction.

(1) Climb, cruise, and descent flight regimes. The more critical of the following should be induced into the automatic pilot system.

(i) A signal about any axis equivalent to the cumulative effect of any single failure, including autotrim (if installed).

(ii) The combined signals about all affected axes, if multiple axes failures can result from the malfunction of any single component.

(2) Limit Loads. The simulated failure and the subsequent corrective action should not create loads in excess of structural limits or result in dangerous dynamic conditions or deviations from the flight path. Additional guidance regarding the method of determining pilot recognition times and reasonable flight path deviation due to these simulated failures is contained in paragraph AC 29 Appendix B b(6). Resultant flight loads outside the envelope of zero to 2g will be acceptable provided adequate analysis and flight test measurements are conducted to establish that no resultant aircraft load is beyond limit loads for the structure, including a critical assessment and consideration of the effects of structural loading parameter variations (i.e., center of gravity, load distribution, control system variations, maneuvering gradients, etc.). Analysis alone may be used to establish that limit loads are not exceeded where the aircraft loads are in the linear range of loading (i.e., aerodynamic coefficients for the flight condition are adequately established and no significant nonlinear air loadings exist). If significant nonlinear effects could exist, flight load survey measurements may be necessary to substantiate that the limit loads are not exceeded. The power for climb should be the most critical of: (1) that used in the performance climb demonstrations; (2) that used in the longitudinal stability tests; or (3) that actually used for operational climb speeds. The altitude loss should be measured.

(3) Maneuvering Flight. Malfunctions should also be induced into the automatic pilot system similar to paragraph c(1). When corrective action is taken, the resultant loads and speeds should not exceed the values contained in paragraph c(2). Maneuvering flight tests should include turns with the malfunction induced when maximum bank angles for normal operation of the system have been established and in the critical aircraft configuration and/or stages of flight likely to be encountered when using the automatic pilot. The altitude loss should be measured.

(4) Oscillatory Tests.

(i) An investigation should be made to determine the effects of an oscillatory signal of sufficient amplitude to saturate the servo amplifier of each device that can move a control. The investigation should cover the range of frequencies that can be induced by a malfunction of the automatic pilot system and systems functionally connected to it, including an open circuit in a feedback loop.

(ii) The results of this investigation should show that the peak loads imposed on the parts of the aircraft by the application of the oscillatory signal are within the limit loads for these parts.

(iii) The investigation may be accomplished largely through analysis with sufficient flight data to verify the analytical studies or largely through flight tests with analytical studies extending the flight data to the conditions which impose the highest percentage of limit load to the parts.

(iv) When flight tests are conducted in which the signal frequency is continuously swept through a range, the rate of frequency change should be slow enough to permit determining the amplitude of response of any part under steady frequency oscillation at any critical frequency within the test range.

(5) Recovery of Flight Control. To aid in recovery of the rotorcraft, after a malfunction occurs, one pilot should be able to physically overpower the autopilot and then disengage it with ease, and it should remain disengaged until further pilot action to reengage. The control to disconnect the autopilot should be easily available to the pilot who is now resisting the malfunctioning force of the autopilot. It is recommended that the disconnect button be placed on the cyclic control. It should be red and conspicuously marked “Autopilot Disconnect.” The pilot should be able to return the rotorcraft to its normal flight attitude under full manual control without exceeding the loads or speed limits defined in paragraph c(2) and without engaging in any dangerous maneuvers during recovery. The maximum servo authority used for these tests should not exceed those values shown to be within the structural limits for which the rotorcraft was designed. The maximum altitude loss experienced during these tests should be measured.

(6) External Interfaces. The autopilot system should have appropriate interlocks to its engagement to ensure it does not operate improperly as a result of information furnished by an external device or system. An example of this is the navigation receivers and the compass system. If for a particular mode of operation the autopilot uses signals from these systems, the autopilot should be interlocked from operating in those modes if invalid information is being received from that system.

d. Automatic Pilot Instrument Approach Approval.

(1) Throughout an approach, no signal or combination of signals simulating the cumulative effect of any single failure or malfunction in the automatic pilot system, except vertical gyro mechanical failures, should provide hazardous deviations from flight path or any degree of loss of control.

(2) The aircraft should be flown down the instrument landing system (ILS) in the configuration and at the approach speed specified by the applicant for approach. Simulated autopilot malfunctions should be induced at critical points along the ILS, taking into consideration all possible variations in autopilot sensitivity and authority. The malfunctions should be induced in each axis. While the pilot may know the purpose of the flight, the pilot should not be informed when a malfunction is about to be or has been applied except through aircraft action, control movement, or other acceptable warning devices.

(3) An engine failure during an automatic ILS approach should not cause a lateral deviation of the aircraft from the flight path at a rate greater than 3° per second or produce hazardous attitudes.

(4) If approval is sought for ILS approaches initiated with one engine inoperative, the automatic pilot should be capable of conducting the approach.

(5) Deviations from the ILS flight profile should be evaluated as follows:

(i) The rotorcraft should be instrumented so the following information is recorded—

(A) The path of the rotorcraft with respect to the normal glide path;

(B) The point along the glide path when the simulated malfunction is induced;

(C) The point where the pilot indicates recognition of the malfunction; and

(D) The point along the path of the rotorcraft where recovery action is initiated.

(ii) Data obtained from the point of the indicated malfunction to the point where the rotorcraft has either again intersected the glide slope or is in level flight will define the deviation profile. When changes to the aircraft autopilot configuration are made during the approach and these changes alter the deviation profile, additional data should be obtained to define each of the applicable deviation profiles. An example of a deviation profile is shown in figure AC 29.1329-1.

(iii) Recoveries from malfunctions should simulate under-the-hood instrument conditions with an appropriate time delay between pilot recognition of the fault and initiation of the recovery at all altitudes down to 80 percent of the minimum decision altitude for which the applicant requests approval.

(iv) Recoveries from malfunctions at altitudes between 80 percent of the minimum decision altitude for which the applicant requests approval and the minimum altitude for which the applicant requests approval to operate the autopilot may be visual with no time delay between pilot recognition of fault and initiation of recovery.

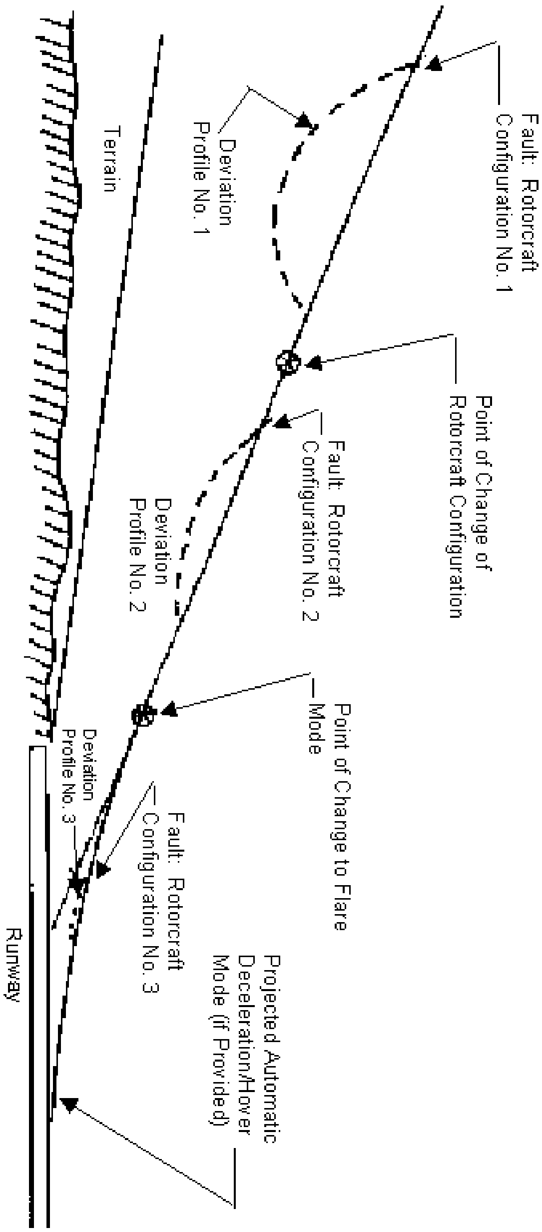
(v) The minimum altitude at which the autopilot may be used should be determined as the altitude that results in the critical deviation profile becoming tangent with a minimum operational tolerance line. An example of this may be found in figure AC 29.1329-2. The 29:1 slope of the minimum operational tolerance line provides a 1 percent gradient factor of safety over the 50:1 obstacle clearance line. An additional factor of safety is provided by measuring the 29:1 slope from the horizontal at a point 15 feet above the runway threshold. It is recognized that this minimum altitude will vary with glide slope angle. Information regarding these variations should be obtained and presented.

(vi) A malfunction of the autopilot during a coupled ILS approach should not place the aircraft in an attitude that would preclude conducting a satisfactory go-around or landing.

e. Servo Authority. The automatic pilot system should be installed and adjusted so the system tolerances established during certification tests can be maintained in normal operation. This may be ensured by conducting flight tests at the extremes of the tolerances. Those tests conducted to determine that the automatic pilot system will adequately control the aircraft should establish the lower limit. Those tests to determine that the automatic pilot will not impose dangerous loads or deviation from the flight path should be conducted at the upper limit. Appropriate aircraft loadings to produce the critical results should be used.

f. Rotorcraft Flight Manual Information. The following information should be placed in the rotorcraft flight manual:

(1) In the Operating Limitations Section. Airspeed and other applicable operating limitations for use of the autopilot.



NOTE: Point of change of rotorcraft configuration may be more than one point. For instance:

1. Gain changes along the glide path.
2. The 200 ft. or middle marker transition.

Figure AC 29.1329-1 Deviation Profile

(2) In the Operating Procedures Section. The normal operation information.

(3) In the Emergency Operation Procedures Section.

(i) A statement of the downward flight path deviation in the cruise, climb, and descent configurations and the maneuvering flight configuration in accordance with paragraphs d(5)(iii) and d(5)(iv) of this paragraph, if this deviation exceeds 100 feet.

(ii) True profiles of deviations below the glide slope or projected flare path for the critical conditions tested in accordance with paragraphs d(5)(iv) (see figure AC 29.1329-1) and the deviation profile indicating the lowest altitude at which the autopilot can be used, as referenced in paragraph d(5)(v), if applicable, and if this deviation exceeds 100 feet or excessive deviation for an ILS approach.

g. There should be a means of sequencing actions or interlocking engagement with sensor inputs to prevent autopilot initiated maneuvers that could result in hazardous operations due to:

(1) Engagement of the autopilot;

(2) Malfunctions of autopilot input or feedback signals that could result in unbounded output commands.

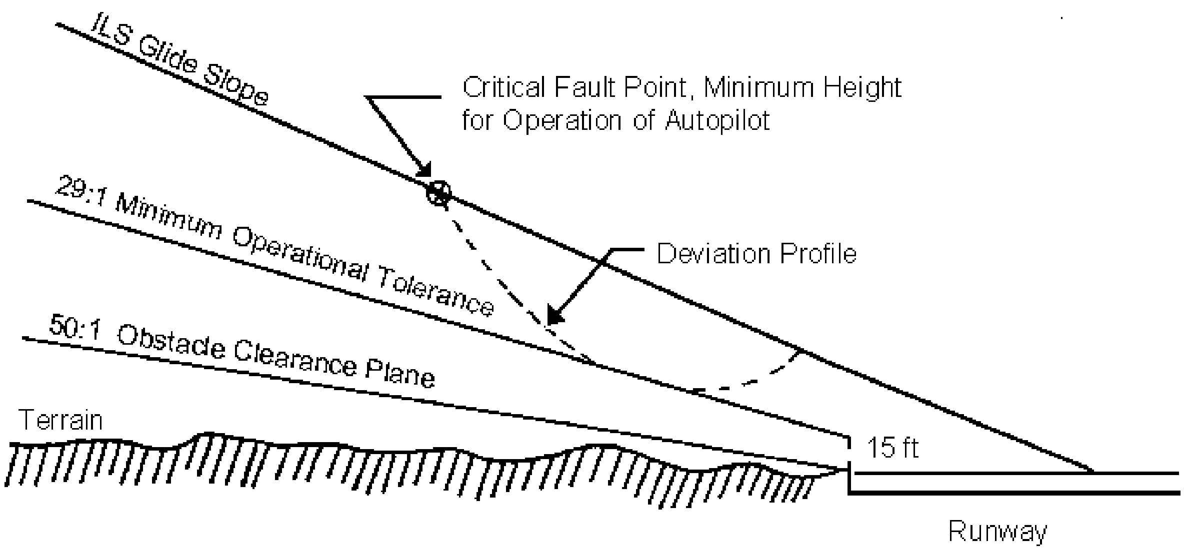


FIGURE AC 29.1329-2 OPERATIONAL LIMITATION

AC 29.1331. § 29.1331 (Amendment 29-24) INSTRUCTIONS USING A POWER SUPPLY.

a. Explanation. The rule concerns each flight instrument using a power supply that is installed in a Category A rotorcraft. A reference to paragraph AC 29.1303 will give a listing of the flight instruments that are specifically required for certification. The discussion included in this paragraph is directed toward electrical instruments since these are the type normally installed in Category A rotorcraft. It should be noted, however, that the rule is not restricted to electrical instruments. Further, the discussion provided here can be used to evaluate non-electrical applications.

b. Procedures.

(1) This requirement must be considered when designing the electrical distribution system for Category A rotorcraft. It states that each required flight instrument must have two independent sources of power and a means of selecting either source. The flight instruments required for certification are listed in paragraph AC 29.1303, and independent power sources are discussed in paragraph AC 29.1355b(4).

(2) Some older flight instruments may not have integral visual means to indicate that adequate power is being supplied to the instrument as required by the rule. For these instruments, external annunciation has been accepted that monitors the presence of adequate voltage at the power pin on the electrical disconnect that mates with the electrical connector on the back side of the instrument. The annunciator light should be located in close proximity to the indicator and placarded to identify its function. Note that the rule requires the voltage monitored to be within the approved limits for the instrument to be adequate. Since relay coils normally operate well outside the approved instrument voltage limits, the use of a relay contact that closes when the monitored voltage drops low enough to pull in or release a relay coil does not normally result in a satisfactory design to meet the regulatory requirement. Annunciator lights provided for this application are normally red.

(3) The power supply system requirements of this rule should be coordinated with the requirements of § 29.1355 (see paragraph AC 29.1355). Both rules concern equipment or systems that require two independent sources of electrical power. Examples of faults in the distribution system to be considered include open feeders, shorted feeders, shorted busses, etc.

(4) Amendment 29-24 revised the regulation to further clarify the power adequacy indication requirement. The clarification provided was intended to make it easier to understand the meaning of adequate power in the event it was necessary to provide separate annunciation. The application of the rule in each form (before and after Amendment 29-24) should be the same.

AC 29.1333. § 29.1333 (Amendment 29-24) INSTRUMENT SYSTEMS.

a. Explanation. Prior to Amendment 29-24, this requirement was titled "Duplicate Instrument Systems," and its provisions were intended to be applied when duplicate flight instruments were required by any operating rule. Due to the increased complexity of instrumentation that is available and being used, it was considered appropriate to revise the provisions of this requirement to more appropriately consider the extreme range of operation environments to which rotorcraft are now routinely exposed. It is the intent of this rule to prevent degrading of the first pilot's instrument system, or the only pilot's instrument system in a single-pilot-approved rotorcraft, by not permitting peripheral systems to be connected to it. In addition, equipment must not be connected to operating systems for the second pilot's required instruments unless it is extremely improbable that failure of such additional equipment would affect that operating system. Similar provisions are also included in Appendix B to Part 29, Airworthiness Criteria for Helicopter Instrument Flight.

b. Procedures.

(1) The provisions of the current rule are essentially self-explanatory.

(2) If the certification basis of the rotorcraft is prior to Amendment 29-24, the provisions are more precise; however, they only apply in the instance where duplicate instruments are required by the operating rules.

(3) If an IFR approval is part of the certification effort, then Part 29, Appendix B, applies, and the provisions of paragraph VIII(b)(5) are essentially the same as the current rule. If the certification basis of the rotorcraft is prior to Amendment 29-24, and an IFR approval is being added, the instrument systems should be carefully reviewed since their design may not have considered the provisions of the IFR rule.

AC 29.1335. § 29.1335 (Amendment 29-14) FLIGHT DIRECTOR SYSTEMS.

a. Explanation. This section prescribes the accepted display criteria for a rotorcraft three-cue flight director providing command guidance for pitch, roll, and power. Three-cue flight directors for rotorcraft use the usual pitch and roll command cues with the third cue displayed on the left side of the attitude director indicator (ADI). These instruments can be used in either the two-axes or three-axes modes. In either mode, the lateral command cue controls the roll attitude, and the vertical command cue controls the pitch attitude. The rotorcraft attitude, controlled by the cyclic control, is changed to satisfy the flight director commands. The third cue, when displayed, commands collective pitch position and is used when an airspeed or pitch attitude mode and a vertical mode (altitude hold, glide slope, etc.) are selected.

(1) The general convention for flight director design is that each command bar is a "fly to" command. The motion of the flight director indicator is such to command a

corresponding sense of control system motion. This is true of flight director pitch and roll commands and should hold true for additional commands such as collective pitch.

(2) Some consideration should be given to the collective, or third cue, display. For example, if the collective symbol is selected as the fixed index, the command cue and collective pitch control should move in opposite directions when collective pitch changes are made. This configuration would constitute a conventional “fly to” indicator. If the collective symbol is selected for the movable index, the direction of motion of the collective symbol will coincide with the direction of collective pitch changes. In this case the moving collective symbol does not comply with the “fly to” convention; however, this configuration has been approved by the FAA/AUTHORITY with special symbology, special background effects, and special color coding, and has performed satisfactorily in service.

b. Procedures. The recommended display for a three-cue flight director incorporates the standard pitch and roll command symbols, either pitch and roll bars or the “V” bar display. The third cue, or collective symbol, should be located on the left side of the ADI. The shape of the moving cue and the background display should be unique to avoid being confused with a glide slope display or angle of attack display. One display uses a third cue, shaped like a small handle, to aid in identifying it as the collective pitch symbol.

(1) The color of the pitch and roll command indicators, the aircraft symbol, the background marking of the third cue, and third cue itself, should be consistent. The optimum color scheme uses the same color for the aircraft symbol and the collective symbol. This is usually fire orange. The command cues including the collective cue also should use the same color, usually yellow. The rationale for the different colors is that the aircraft symbol and the collective symbol (the same color) are moved toward their respective command cues. If the pitch command cue is above the center, the aircraft symbol is raised (nose pulled up) and, if the collective command cue is above the collective symbol, the collective pitch is raised, moving the collective symbol towards the command cue.

(2) If the attitude director indicator (ADI) provides a monochromatic display, the collective pitch cue and its background markings must be distinctive to reduce the chance of being confused with the glide slope indicator. This can be accomplished through the use of different shaped cues and background marks. A round cue with a chevron-shaped background marking has been satisfactory.

AC 29.1337. § 29.1337 (Amendment 29-13) POWERPLANT INSTRUMENTS - (Paragraph (b) - FUEL GAUGE CALIBRATION).

a. Explanation. Section 29.1337(b) requires, in part, a means to indicate to the flightcrew the quantity of useable fuel in each tank in flight. Since the flight attitude of a rotorcraft may vary significantly with CG (center of gravity) and airspeed, a standard attitude for calibration of the fuel quantity gauge is needed. In addition, guidelines for gauge accuracy and comments regarding other fuel quantity gauging aspects are offered.

b. Procedures.

(1) Determine the rotorcraft pitch attitudes for most forward and most aft CG at a median gross weight and at an airspeed of 0.9 V_{NE} or 0.9 V_H, whichever is less. The mean attitude of the extremes defined above, further adjusted for lateral CG effects, if necessary, define the rotorcraft attitude for fuel gauge calibration.

(2) After establishing the calibration attitude, the requirements of § 29.1337(b) can be accomplished. The aircraft should be placed in the calibration attitude. Add fuel to the filler neck spillover level. Defuel the aircraft in increments corresponding to fuel gauge increment markings or at least 10 increments until gauge zero is obtained. Precautions should be taken during this step to be sure that the fuel transmitter is sensing fuel level and not simply reflecting a physical “STOP” or end point in the system range. The fuel remaining in the tank below the “ZERO” mark must not be less than that amount determined by flight testing under § 29.959. (Otherwise, the zero point must be adjusted upward.) The gauging system accuracy is acceptable when it meets a tolerance of ±2 percent of the total useable fuel plus ±4 percent of the remaining usable fuel at any gauge reading, provided that the gauge indicates zero fuel with unusable fuel in accordance with § 29.959 in the tank. (For a 100-gallon tank this formula would allow a ±6-gallon error at the full level, ±4-gallon error at 50-gallon level, converging to a ±2-gallon error at low fuel with the further provision that the zero mark accurately reflects unusable fuel.)

(3) Certain other aspects of a fuel gauging system need attention in order to minimize fuel exhaustion incidents:

(i) Gauge reading with the aircraft at ground attitude is frequently used by the crew in calculating range, weight and balance, and actual gross weight. Significant gauge errors in either direction during this reading can introduce hazards to the operation of the aircraft. If a calibration at this attitude indicates an unconservative error in excess of 6 percent of the gauge reading, corrective information should be applied adjacent to the fuel quantity gauge or be made available to the crew in other handbook data.

(ii) Flight during hover with maximum rearward wind may introduce significantly different fuel gauge readings. A check should be made to assure that the gauge is either accurate or at least does not read high (unconservative) in this attitude.

(4) Fuel gauging system transmitters which are strictly volumetric measuring devices (float-actuated variable rheostats) introduce a gauge readout error of about 5 percent if calibrated with a fuel temperature of 0° C and subjected to -55° C fuel or +55° C fuel. This error may be minimized by calibrating the gauge with fuel temperature in the middle of the useful range; i.e., 15° C.

(5) Capacitance transmitters have become the standard for most modern fuel systems. These transmitters ordinarily need no temperature compensation since the fuel volume and the fuel dielectric constant vary inversely as temperature changes. The basic capacitance transmitter does not compensate for the different dielectric values to be expected with different type fuels. An add-on capacitance located so as to be submerged in fuel at all times can be devised to automatically compensate for other fuels.

AC 29.1337A. § 29.1337 (Amendment 29-26) POWERPLANT INSTRUMENTS.

a. Explanation. Amendment 29-26 adds § 29.1337(e) that requires certain rotor drive system transmissions and gearboxes to be equipped with chip detector systems. These detectors will sense and signal the presence of ferromagnetic particles to the flight crew. The rule also requires a means to permit the crewmembers to check, in flight, the function of each detector’s electrical circuit and signal. This amendment will improve the level of safety available with the installation of chip detector systems.

b. Procedures. All of the policy material pertaining to this section remains in effect. Additionally, the following information is added about chip detectors. The chip detectors should:

(1) Indicate the presence of ferromagnetic particles in the transmission or gearbox;

(2) Be easily removable for inspection of the magnetic poles for metallic chips; and,

(3) Prevent the loss of lubricant in the event of failure of the retention device for the removable portion of the chip detector (debris monitor).

(4) Provide a test system to allow the crew to check, in flight, the function of each detector and wiring. The test circuit should test, at least, as much of the circuitry as reasonably possible. Where detectors are used that have a test feature in the form of an extra pin, all of the circuit, exclusive of the detector may be tested. Some chip detectors have a fuzz burner capability to eliminate nuisance indication of non-relevant conducting materials that result from oil contamination and very small wear particles.

SUBPART F - EQUIPMENT

ELECTRICAL SYSTEMS AND EQUIPMENT

AC 29.1351. § 29.1351 (Amendment 29-40) ELECTRICAL SYSTEMS AND EQUIPMENT -GENERAL.

a. Explanation. With the advent of more sophisticated rotorcraft and operations under more critical conditions, such as IFR and icing, it is essential that the electrical system be very carefully analyzed and evaluated to assure proper operation under any foreseeable operating condition, and that hazards do not result from any malfunctions or failures.

b. Procedures.

(1) An acceptable method of preparing an electrical load analysis is given by Military Specification MIL-E-7016F, and use of this standard is preferred since it has been received widespread acceptance. If other formats have been used and have been considered acceptable, their continued use is encouraged.

(2) Generating systems must be analyzed, inspected, or tested to assure conformance to the following criteria.

(i) For Category A, the generating system must perform as specified in § 29.1309(d) and (e).

(ii) No probable malfunction in the generating system or in the generator drive system may result in the permanent loss of service to electric utilization systems, which are necessary to maintain controlled flight and to effect a safe landing, unless the aircraft is equipped with an independent source of electrical power capable of supplying continuous emergency service to these utilization systems. A probable malfunction is any single electrical or mechanical malfunction or failure which is considered probable on the basis of past service experience with similar components in aircraft applications. This definition should be extended to multiple malfunctions when:

(A) The first malfunction would not be detected during normal operation of the system, including periodic checks established at intervals which are consistent with the degree of hazard involved; or

(B) The first malfunction would inevitably lead to other malfunctions.

(3) The generator drive system includes the prime movers (propulsion engines or other) and coupling devices such as gear boxes or constant speed drives.

(4) An electric utilization system is a system of electric equipment, devices, and connected wiring which utilizes electric energy to perform a specific aircraft function.

(5) The specific electric utilization systems, which are necessary to maintain controlled flight and effect a safe landing, will vary with the type of aircraft and with the nature of the operation in which the aircraft is utilized. Examples of systems which may be in this category are as follows: basic flight instruments, minimum navigation equipment, minimum radio communications, and control system boost.

(6) Where crew corrective action is necessary,

(i) Adequate warning should be provided for any malfunction or failure requiring such corrective action.

(ii) Controls should be so located as to permit such corrective action during any probable flight situation.

(iii) If corrective action must be taken within a specified time interval for continued safe operation of the generating system, it should be demonstrated that such corrective action can be accomplished within the specified time interval during any probable flight situation. For Category A rotorcraft, compliance with § 29.903(b)(2) must be considered.

(iv) The procedure to be followed by the crew should be detailed in the Rotorcraft Flight Manual.

(7) Voltage and current supplied by each generator are considered essential parameters for definition of system operation and most systems are provided with voltmeters and ammeters to display these parameters to the crew. Some recent designs have annunciated safe operation of each generator with lights and have eliminated the voltmeter and ammeter. For these systems, in addition to distribution system design precautions, parameters such as over and under voltage, reverse current sensing, feeder ground faults, and over and under frequency (AC generators only) are being monitored and provided as inputs to the generator annunciators. For systems not incorporating voltmeters and ammeters, and with automatic protective switching and annunciator lights, the pilot should be provided as a minimum, with sufficient information to determine the type of fault, and to identify portions of the system that have been lost. If additional limitations such as maximum loading of portions of the system are necessary to account for fault conditions, that information should be made available to appropriate personnel (crew, owner, modifier, etc.) to assure the limits are not exceeded.

(8) For rotorcraft with a certification basis of FAR 29 after Amendment 29-14 (effective on July 18, 1977), the electrical wire and cable insulation and other materials used to show compliance with § 29.1351(d) must be self-extinguishing when tested in accordance with Part 25 Appendix F. This means the wire must be tested at an angle

of 60° in accordance with the applicable portions of Appendix F of FAR 25 which contain acceptable test procedures and define burn length.

(9) An area where a possibly hazardous malfunction of an electrical power source might occur is the supply of cooling air to the electrical generators. The hazard exists because the failure of a generator bearing usually produces metallic sparks and hot surfaces which are a potential ignition source. Consideration should be given to this failure. One method is if the generator is rated explosion proof, then the intake and output of cooling air into the engine compartment should not cause a hazard. If the generator is not rated explosion-proof and a failed bearing test cannot conclusively demonstrate that failure of the generator will not produce an ignition source, then cooling air should be ducted into and out of the generator from outside the aircraft. The ducting material should be sufficient to contain the failed generator fragments.

(10) Generator ratings are often the result of installation temperature limitations. The determination of these limitations, if any, is by testing the actual installation. The procedures for performing generator cooling tests are as follows:

(i) Test Requirements.

(A) General. The applicant should contact the generator (alternator) manufacturer and obtain the maximum limits for the unit to be tested. This will normally be in terms of temperatures at various locations within the unit (stator, bearings, diodes, heat sinks, brushes, etc.) or in terms of pressure drop across the generator. The manufacturer should either supply an instrumented unit or give complete details for instrumenting the test unit.

(B) Instrumentation.

(1) Load Bank. A load bank will usually be necessary to load the test unit to the amperage limit for which approval is requested.

(2) Ammeter. An ammeter should be provided with sufficient resolution to assure the amperage load is being maintained at the desired level.

(3) Temperature/Pressure Readouts. Readouts which are compatible with the temperature or pressure sensors installed in the test unit should be provided.

(4) Calibration Records. Calibration records should be available for all instrumentation.

(5) Recordings. Permanent recordings should be provided for time, temperatures, current and/or pressure. The recording device should have provisions for placing event marks on the recording medium.

(C) Regulatory References. Sections 29.1301, 29.1309, 29.1351, 29.1363(b), 29.1521(e), 29.1041, 29.1043, 29.1045, 29.1047, and 29.1049.

(D) Miscellaneous. The results obtained from the tests should be corrected for hot day conditions using a standard lapse rate (3.6° F/1,000 feet). The tests are conducted to determine the maximum generator capacity that does not result in surpassing the limits given from the manufacturer. This is for a continuous rating, any credit for short time over current ratings must also be verified by the same methods, particularly for short time ratings longer than 5 seconds.

(ii) Test Procedures.

(A) Single Engine Procedure.

(1) The cooling test is to be conducted during ground operation, climb-out, cruise, approach, and hover flight regimes.

(2) All ground operational and in-ground effect hover tests should be conducted in ambient winds of 5 knots or less. Wind direction relative to the aircraft should be from the most critical direction.

(3) The battery may be connected to the bus during the generator/alternator cooling test. The generator/alternator temperatures should be recorded at intervals sufficiently close to show the rate of temperature increase and stabilization. The temperature may be considered stabilized when it peaks and has not increased in the last 5 minutes. The climb-out speed and power setting should correspond to the best rate of climb speed, using maximum continuous power or any other normal conditions of climb that would cause the generator/alternator temperatures to be critical. The cruise test should be conducted at maximum altitude in the cruise configuration. Generator/alternator cooling should be conducted at rated output consistent with the RPM at which it is operating. For instance, during the ground tests the engine RPM may be lower than that necessary to sustain maximum rated amperage output. In this case the maximum amperage output of the generator/alternator corresponding to the lower RPM should be assured.

(4) The test sequence should begin with about 30 minutes of ground operation to account for taxi and holding times, and end 5 minutes after all temperatures have peaked after engine shut down.

(B) Multi-engine Procedures. Conduct a generator cooling test in accordance with the following procedure:

(1) All ground operational and in-ground effect hover tests should be conducted in ambient winds of 5 knots or less. Wind direction relative to the aircraft should be from the most critical direction.

(2) After engine start, load the instrumented generator to its proposed amperage limit and begin recording temperatures.

(3) A total of 30 minutes should be spent on the ground prior to takeoff. This is to account for taxi and holding times.

(4) After takeoff, climb at single-engine best-rate-of-climb speed using maximum continuous power, to the single-engine service ceiling. Above this, continue at twin-engine best-rate-of-climb speed, using maximum continuous power on both engines, to maximum altitude.

(5) Cruise at maximum altitude until all generator temperatures stabilize. Temperatures shall be considered stabilized when they have peaked and have not increased for a period of 5 minutes.

(6) Descend, conduct an approach to include a go-around, hover until temperature stabilizes, then land and continue to record temperatures after shut-down until 5 minutes after all temperatures have peaked.

(7) Conduct cooling tests with the rotorcraft hovering at both the minimum and maximum hover altitudes.

(8) Correct all results for hot day conditions. Use the standard lapse rate of 3.6° F/1,000 feet for consideration of altitude. See paragraph AC 29.1309b(2)(i) for details on temperature correction.

(C) Manufacturer’s Limits. If at any time during the testing it appears the manufacturer’s limits are to be exceeded, the amperage load on the test generator/alternator should be reduced to prevent this from happening.

(D) Miscellaneous. The results obtained from the tests should be corrected for hot day conditions using a standard lapse rate (3.6° F/1,000 feet). The tests are conducted to determine the maximum generator capacity that does not result in surpassing the limits given from the manufacturer. This is for a continuous rating; any credit for short time over current ratings must also be verified by the same methods, particularly for short time ratings longer than 5 seconds.

c. Operation with normal electrical power generating system inoperative. See FAR 29.1351(d).

(1) Definition: Normal electrical power generating system. The term normal electrical power generating system is intended to include all electrical power sources used for operation of the rotorcraft under any approved normal operating condition (VFR, IFR, Icing, etc.), not including batteries and emergency electrical power sources.

(2) All rotorcraft (See FAR 29.1351(d)(1) Amendment 29-40).

(i) FAR 29.1351(d)(1) requires, for all rotorcraft, continued safe VFR operation for a period of at least 5 minutes with the normal electrical power system inoperative. If loss of the normal electrical power generating system, followed by depletion of battery power, could prevent safe flight and landing, adequate warning of loss of the normal electrical power generating system should be provided for compliance with FAR 29.1309(c), and Flight Manual procedures compatible with the available battery endurance should be provided.

(ii) One possible cause of loss of the normal electrical power generating system is engine failure. The requirement specifies consideration of engine flameout and restart attempts. A minimum battery endurance of 5 minutes is specified. To ensure safe operation under all conditions, however, the battery endurance should be not less than the time required for an autorotative descent to sea level from the maximum operating altitude. Where applicable, allowance should be made for the use of the batteries for attempts to restart the engines during the descent. It may be necessary to include limitations on the number of attempted starts or to provide a separate dedicated battery for such purposes.

(3) Category A rotorcraft (See FAR 29.1351(d)(2) Amendment 29-40).

(i) FAR 29.1351(d)(2) is applicable to Category A rotorcraft and requires that provision be made to ensure adequate electrical supplies to those systems which are necessary for continued safe flight and landing in the event of a failure of all normal generated electrical power. All components and wiring of the alternate supplies should be physically and electrically segregated from the normal system and should be such that no single failure, including the effects of fire, the cutting of a cable bundle, or the loss of a junction box or control panel will affect both normal and alternate supplies.

(ii) In considering the systems which should remain available following the loss of the normal electrical power generating system, consideration should be given to the role and flight conditions of the rotorcraft and the possible duration of flight time to reach a suitable landing site and make a safe landing.

(iii) The systems required by FAR 29.1351(d)(2) may differ between rotorcraft types and roles and should be agreed with the Authority. They should normally include:

(A) Attitude information;

(B) Radio communication and flight crew intercommunication;

(C) Navigation;

(D) Cockpit and instrument lighting;

(E) Heading, airspeed and altitude information, including appropriate pitot head heating;

(F) Adequate flight controls;

(G) Adequate engine instrumentation and control;

(H) Such warnings, cautions, and indications as are required for continued safe flight and landing;

(I) Any other services required for continued safe flight and landing; e.g. fire extinguishing, emergency flotation equipment, landing light.

(iv) Emergency Power Source Duration and Integrity

(A) Time Limited Power Source. Where an emergency power source provided to comply with FAR 29.1351(d)(2) is time limited (e.g., battery), the required duration will depend on the type and role of the rotorcraft. Unless it can be shown that a lesser time is adequate, such a power source should have an endurance of at least half the rotorcraft endurance, or the Flight Manual limitations section should define aircraft endurance. However, an endurance of less than 30 minutes would not normally be acceptable. The endurance, with any associated procedures, should be specified in the Flight Manual. The endurance time should be determined by calculation or test, due allowance being made for-

(1) Delays in flight crew recognition of failures and completion of the appropriate drills where flight crew action is necessary. This should be assumed to be 5 minutes provided that the failure warning system has clear and unambiguous attention-getting characteristics and where such a delay is compatible with the crew's primary attention being given to the corresponding emergency procedures and/or other possible related failures such as engine fire, fumes in the cockpit, etc. A delay of less than 5 minutes may be acceptable if justified by simple procedures or an adequate degree of automation.

(2) The minimum voltage acceptable for the required loads, the battery state of charge, the minimum capacity permitted during service life and the battery efficiency at the discharge rates and temperatures likely to be experienced. Unless otherwise agreed for the purpose of this calculation, a battery capacity at normal ambient conditions of 80 percent of the nameplate rated capacity, at the 1 hour rate, and a 90 percent state of charge, may be assumed (i.e., 72 percent of nominal demonstrated rated capacity at +20° C).

(3) For those rotorcraft where the battery is also used for engine or APU starting on the ground, it should be shown that following engine starts, the charge rate of the battery is such that the battery is maintained in a state of charge that will ensure

adequate emergency power source duration should a failure of generated power occur shortly after takeoff.

NOTE: This could, for example, be achieved by ensuring that, following battery-powered starting, the battery charging current has fallen to a specified level prior to take-off.

(B) Non-time-limited Power Source. Where an emergency power source is provided by a non-time-limited source (e.g., standby generator driven by APU, transmission, pneumatic or hydraulic motor), due account should be taken of any limitation imposed by rotorcraft speed, altitude, etc., which may affect the capabilities of that power source. In considering the power source, account should be taken of the following:

(1) Auxiliary Power Unit (APU).

(i) An APU capable of continuous operation throughout an adequate flight envelope may be considered an acceptable means of supplying electrical power to the required services provided that its air start capability is adequate and may be guaranteed. Where, however, the APU is dependent for its starting current on a battery source which is supplying critical loads, such starting loads may prejudice the time duration of the flight if APU start is not achieved.

(ii) It may be necessary, therefore, to include limitations on the number of attempted starts or to provide a separate battery for APU starting, if this method of supplying electrical power is adopted. Consideration should also be given to the equipment, services and duration required prior to the APU generator coming on-line. Common failures which could affect the operation of all engines and the APU (e.g., fuel supply) should be taken into consideration.

(2) Transmission-driven Generator.

(i) A transmission-driven generator may be utilized to provide an emergency electrical power source, but due consideration should be given to ensuring that the means of bringing the generator into use are not dependent on a source which may have been lost as a result of the original failure.

(ii) The continuity of electrical power to those services, which should remain operative without crew action prior to the generator being brought into operation, may necessitate the use of a battery, unless the operation of the emergency power source is automatic and immediate in the event of failure of the normal electrical power generating system.

(3) Pneumatic or Hydraulic Motor Driven Power Source. A pneumatic- or hydraulic-motor-driven electrical power source may be utilized subject to the same constraints on activation as the transmission-driven generator (See 3.4.2(b)). Care

should be taken in ensuring that the operation of the pneumatic or hydraulic system is not prejudiced by faults leading to, or resulting from, the original failure, including the loss of, or inability to restart engines.

AC 29.1353. § 29.1353 (Amendment 29-14) ELECTRICAL EQUIPMENT AND INSTALLATIONS.

a. Explanation.

(1) Electrical equipment, controls, and wiring must be installed so that operation of any one unit or system of units will not adversely affect the simultaneous operation of any other electrical unit or system essential to safe operation. Additionally, wiring installation design should be documented sufficiently to maintain configuration control for manufacturing and to assure that the electromagnetic characteristics remain the same as the certification sample.

(2) Results of qualification testing should be available to ensure that the installation of equipment or a system will not result in adverse interference being introduced into the rotorcraft electrical equipment. A good reference for interference testing is the applicable version of Radio Technical Commission of Aeronautics (RTCA) Document No. DO-160, “Environmental Conditions and Test Procedures for Airborne Equipment.”

(3) The DO-160 type tests would normally be accomplished by the equipment manufacturer. The airframe manufacturer’s test are normally more subjective and are oriented more toward watching for unwanted meter movement, noise in the interphone systems, and so forth. The combination of the equipment manufacturer’s tests, supplemented by the airframe manufacturer’s installation tests, should be adequate to assure compliance with this regulation.

b. Procedures.

(1) General. Chapter II of Advisory Circular 43.13-1A, “Acceptable Methods, Techniques, and Practices: Aircraft Inspection and Repair,“ includes considerable guidance regarding the installation of electrical systems (routing, separation, typing, clamping, j-box installations, etc.). The following areas are overlooked in many cases and special emphasis should be placed on them during the compliance inspection of the rotorcraft:

(i) Feeder wires from the rotorcraft’s generators and batteries should be routed separately from utilization system wiring.

(ii) Generator field wiring should be routed separately from generator output wiring. This should begin at the generator and continue to the voltage regulator.

(2) Battery Installation.

(i) As part of the electrical system evaluation, the battery installation should be reviewed to assure the battery is vented and drained. If there is some doubt regarding the ability of the drain to satisfactorily dispose of corrosive fluids, TIA tests should be conducted to resolve the issue. Normally this is done by expelling a dye solution through the drain system during different phases of flight to assure that fluids are drained clear of the rotorcraft. Some aircraft rely on the installation of a sump jar to dispose of corrosive fluids.

(ii) In nickel cadmium batteries are used for engine starts and compliance with § 29.1353(c)(6) is achieved through the use of a temperature monitoring system, the temperature sensor should be located in a position that will most accurately reflect the internal battery temperature without causing adverse effects to the sensor. The location normally used is near the center of the battery. If the sensor is placed between two cells, the indication should be very close to the actual temperature within the cell. If the sensor is placed in a cell strap, there will normally be a period of time just after a heavy current drain (e.g., engine start) when the sensor shows a temperature that is hotter than the actual cell temperature.

(iii) Other aspects of the battery installation can be resolved by reviewing § 29.1353(c), AC 43.13-1A, and AC 43.13-2A, "Acceptable Methods, Techniques, and Practices Aircraft Alterations."

(iv) Battery replacement with other than the original, in electrical systems that are required to have fault clearing features, will require analysis and tests to show that the fault clearing design has not been affected. There are high rate discharge types of batteries that have different internal impedences from slow rate discharge batteries and the reaction in a fault clearing circuit is completely different between the two types. Additionally, the amp-hour rating of nickel cadmium batteries versus lead acid are not the same from a usable aspect. Nickel cadmium batteries voltage discharge curve allows most of the amp hour capacity to be usable from a voltage standpoint, but only about 66% of a lead acid battery amp hour capacity is usable. Any change to the original type design should consider these limitations.

AC 29.1355. <u>§ 29.1355 (Amendment 29-14) DISTRIBUTION SYSTEM</u>.

a. <u>Explanation</u>. None.

b. <u>Procedures</u>.

(1) When determining compliance with the portion of the rule that concerns supplying essential circuits in the event of reasonably probable faults or open circuits, the effects of tripped circuit breakers or blown fuses should be considered.

(2) Various means may be used to ensure an energy supply. Examples include duplicate electrical equipment, throwover switching, and multichannel or loop circuits separately routed.

(3) Essential load circuits are those circuits whose functioning is required to show regulatory compliance with the certification basis. In addition to those circuits specifically required by the regulations, this definition also includes those circuits required by general rules such as § 29.1309.

(4) An independent power source includes not only the electrical power source (e.g., generator) but other items such as a regulator or a reverse current cutout that are necessary to make the electrical power source deliver power to a distribution bus. When a regulatory requirement exists for two independent power sources, the required items should not be shared.

(5) Electrical system faults may occur that will result in a portion of the system (feeders, buses, etc.) being lost. Where portions of the electrical system may be switched from one power source to another to compensate for a fault, it is important that the transfer action not result in the loss of the replacement source. Circuit design should be such as to assure this will not happen.

AC 29.1355A. § 29.1355 (Amendment 29-24) DISTRIBUTION SYSTEM.

a. Explanation. Amendment 29-24 provides clarification for availability of the remaining electrical power source after a failure of one of two independent power sources.

b. Procedures. All of the policy material pertaining to this section remains in effect with the addition that an automatic or manually selectable means is required to maintain operation of the equipment or system for which the two independent power sources were required.

AC 29.1357. § 29.1357 CIRCUIT PROTECTIVE DEVICES.

a. Explanation. Circuit protective devices are normally installed to limit the hazardous consequences of overloaded or faulted circuits. These devices are resettable (circuit breakers) or replaceable (fuses) to permit the crew to restore service when nuisance trips occur or when the abnormal circuit condition can be corrected in flight.

b. Procedures.

(1) Overvoltage protection is specifically required for Category A rotorcraft. For Category B rotorcraft, the possible types of operation should be considered in combination with the presence of an overvoltage condition in the generating system. The regulatory requirement to support this assumption is § 29.1309(b). If the presence

of an overvoltage condition in the electrical system will not cause a hazard to the rotorcraft, the electrical system could be approved for Category B without overvoltage protection. If a hazardous condition will result from the overvoltage condition, then overvoltage protection must be provided.

(2) Automatic reset circuit breakers, which automatically reset themselves periodically, should not be applied as circuit protective devices. If an abnormal circuit condition cannot be corrected in flight, the decision to restore power to the circuit involves a careful analysis of the flight situation. The necessity of the circuit for continued safe flight should be weighed against the hazards of resetting on a possibly faulted circuit. Such an evaluation is properly an aircraft crew function which cannot be performed by automatic reset circuit breakers. To assure crew supervision over the reset operation, circuit protective devices should be of such design that a manual operation is required to restore service after tripping. Circuit breakers must be designed such that the tripping mechanism cannot be overridden by the operating control, and these circuit breakers are known as the “trip free” type.

(3) Automatic reset circuit breakers may be used as integral protectors for electrical equipment (e.g., thermal cutouts) provided that circuit protection is also installed to protect the cable to the equipment.

(4) If the installation of a system is required as a prerequisite to showing compliance with the regulations, it is generally considered to be essential to some phase of flight or it would not be required. It follows from this that the circuit protective device associated with those systems is generally considered to be essential for safety in flight and should therefore be accessible to the crew in the cockpit. This includes the basic electrical system, the distribution system, and utilization systems that are required. Some examples of required utilization systems are those specified by §§ 29.1303, 29.1305, 29.1307, 29.1381, 29.1383, 29.1385, 29.1401, and 29.1431. Where continued safe flight to the destination is considered to be sufficiently assured, certain required circuits have been excepted from being accessible to the crew in the cockpit. Voltmeter and ammeter circuit protective devices are examples of ones that have been excepted. Some utilization systems, although not specifically required by FAR 29, may be required because of the particular design presented for certification. Circuit protective devices for systems in this category are considered to be required and must be accessible.

(5) The following are considered to be acceptable compliance with the “readily reset” provision of § 29.1357(d).

(i) For a crew of two pilots, it is satisfactory for one of the crewmembers to move his seat and loosen his shoulder harness in order to properly identify and reset or replace a circuit protective device. It is not satisfactory for one of the crewmembers to leave his crew station to reset the circuit protective device.

(ii) For a single pilot situation, with the seat belt and shoulder harness normally adjusted, the circuit protective device location should allow for identification of the opened circuit protector and reset capability while the pilot is flying the rotorcraft.

(6) If fuses are used, there should be spare fuses for use in flight equal to at least 50 percent of the number of fuses for each rating required for complete circuit protection. This only applies to fuses used to protect systems that are required to show compliance with the regulations. Spare provisions need not be made for nonrequired convenience type installations although it is encouraged. The spare fuses should be stored in a location where they are readily accessible to the crew. If not directly visible to the crew, information regarding location of the spare fuses should be provided. One acceptable location is on the fuse panel in a holder with no wire terminations and identified "spare" with the "size."

(7) Refer to paragraph AC 29 MG 2 for specific tests of circuit protection for the total electrical system.

AC 29.1357A. § 29.1357 (Amendment 29-24) CIRCUIT PROTECTIVE DEVICES.

a. Explanation. Amendment 29-24 to the regulations provides the requirements for automatic reset circuit breakers and expands the requirements for disconnecting power sources and transmission equipment to include other malfunctions besides overvoltage. The overvoltage protection requirements are extended to both Category A and Category B rotorcraft. Clarification was added to the requirement for each essential load to have individual circuit protection.

b. Procedures.

(1) All of the policy material pertaining to this section remains in effect except that protection from hazardous overvoltage and other malfunctions that would damage equipment should be provided for both Category A and Category B rotorcraft. The protective sensing/switching devices should disconnect the overvoltage or other malfunctions with sufficient speed to prevent user equipment damage.

(2) In addition, each essential load should have individual circuit protection. This generally means each electrical power consuming device should have individual protection. An exception may be simple systems with multiple lights in a single lighting system which would, in most cases, require only one circuit protective device. The decision of whether one or more protective devices are required, is based on how independent each of the loads should be to one another and what the penalty would be if one load faulted and deprived the remaining loads of electrical power.

AC 29.1359. § 29.1359 ELECTRICAL SYSTEM FIRE AND SMOKE PROTECTION.

a. Explanation. This regulation requires that all electrical system components meet the applicable fire and smoke protection provisions of §§ 29.831 and 29.863, and further requires that certain items in designated fire zones must be at least fire resistant. This regulation becomes very significant when failure conditions are considered, and in accordance with the provisions of § 29.831 “reasonably probable failures” must be considered when assuring compliance.

b. Procedures.

(1) When selecting a type of wire, the burning characteristics of that wire are important. Both composition and quantity of resultant smoke and fumes should be considered. The impact of the smoke and fumes on the aircraft cabin occupants should be accounted for.

(2) Wire qualified to MIL-W-25038 is normally used in circuits that “must be at least fire resistant.” Wire qualified to other specifications may also be satisfactory; however, the provisions of the other specifications should be compared to the provisions of MIL-W-25038 to assure the critical areas are not compromised.

(3) Electrical connectors that are located in a designated fire zone and are used in emergency procedures should be at least fire resistant and capable of maintaining the integrity of the circuit. When evaluating these connectors, careful attention should be directed to the entire connector (the contact, the insert, and the shell).

(4) Wire insulated with KAPTON® polyimide film manufactured to MIL-W-81381A, has been used in aeronautical products with varying degrees of success. The U.S. Navy had such a bad service history with KAPTON® insulated interconnect wire in aircraft that in the mid-1980’s the Navy no longer allowed the use of KAPTON® insulated wire. Although the FAA/AUTHORITY has taken no such action, the use of KAPTON® insulated wire requires very special handling. The following areas should be observed when utilizing KAPTON® insulated wire:

(i) The instructions in the KAPTON® wire “Handling Manual” should be strictly followed. This manual may be obtained from E.I. Du Pont de Nemours and Company, Polymer Products Department, Industrial Film Division, Wilmington, Delaware 19898.

(ii) Use in special wind and moisture problem (SWAMP) areas, such as wheel wells, usually requires additional protection for the cable bundles.

(iii) The wire should not be exposed to a combination of either high stress (U.V. or physical) in the presence of water, high humidity, or high pH factor liquids.

(iv) The stiffness and permanent set (memory) of KAPTON® may cause chafing in unrestrained bundles or where KAPTON® insulated wire is bundled with wires of other insulation types.

(v) Care should be exercised in the stripping, stamping, and terminating of KAPTON® insulated wires.

NOTE: KAPTON® is a registered trademark of E.I. Du Pont de Nemours and Company.

AC 29.1363. § 29.1363 ELECTRICAL SYSTEMS TESTS.

a. Explanation. Most of this rule is self-explanatory. Since other regulatory paragraphs also contain requirements regarding functioning and malfunctioning of the electrical system, a recommended test procedure has been included in paragraph AC 29 MG 2 instead of being made a part of this paragraph.

b. Procedures.

(1) Reference paragraph AC 29 MG 2 for a recommended test procedure.

(2) When simulating the electrical characteristics of the distribution system wiring, emphasis should be placed on duplicating the type, gage, and length of the wiring being evaluated. As much as possible, cable bundling and grounding considerations should also be duplicated.

(3) Most laboratory test connected loads would normally be in the form of load banks rather than providing the actual aircraft system. If load banks are used during laboratory testing, additional consideration should be given to these loads when an actual aircraft installation is available.

(4) Limited aircraft testing should also be accomplished to verify that the response of the laboratory drives does adequately simulate the response of the rotorcraft engines under normal and malfunction conditions.

SUBPART F - EQUIPMENT

LIGHTS

AC 29.1381. § 29.1381 INSTRUMENT LIGHTS.

a. Explanation. This section provides minimum performance standards for the instrument lighting system. Section 29.1309(b) is used to evaluate the malfunction aspects of the system. If appropriate, § 29.1309(a) is used to evaluate the equipment under environmental considerations.

b. Procedures.

(1) The overall instrument lighting system should be designed and installed such that single failures that occur will not result in the loss of both primary and secondary (backup) lighting for any instrument or area of the cockpit. In some instances, the system is divided such that the controls for the pilot's panel are separate from the copilot's panel and both of these are separate from the center panel. The ideal is to divide the system such that the impact of single failures will be minimized.

(2) Secondary (backup) instrument lighting should be provided, and this is accomplished in some instances by eyebrow lights. A system that provides general cockpit lighting from a source in the aft area of the cockpit is normally not acceptable since normal positioning and movement of the crew will block this type of light.

(3) The standard does not specify any color requirements for instrument lighting. White is normally provided. The color provided should ensure that the color coding of the instruments is readily identifiable.

(4) The final installed system should be evaluated by a flight test pilot. An actual night flight should be conducted for initial certification of an aircraft. In some instances the vibration characteristics and other flight-induced factors have been demonstrated to seriously affect the pilot's ability to see in the cockpit environment at night. Evaluations following modifications may be conducted with a darkened cockpit on the ground. It should be verified that direct rays are shielded from the pilot's eyes, and that objectionable reflections do not exist. The pilot should also assume failures of various controls, electrical busses, etc., to account for all appropriate failures.

(5) In some instances manufacturers have provided high intensity instrument lighting systems as an option associated with IFR approvals. If provided, this capability should be included in the overall evaluation of the instrument lighting system.

AC 29.1383. § 29.1383 LANDING LIGHTS.

a. Explanation. This section provides minimum performance standards for the installation and normal operation of the landing lights. Certification to this standard is all that is required for approval of the rotorcraft; however, the different operating rules should also be reviewed since they may contain additional requirements. The malfunction considerations are based on the provisions of § 29.1309(b).

b. Procedures.

(1) The performance requirements of this standard are normally evaluated by a flight test pilot, and usually are included in the Type Inspection Authorization as part of the evaluation to be conducted at night.

(2) The installation of the landing light unit(s) should be very carefully evaluated. Many of the units provided are stowed until needed and then driven to their operating position by an electric motor. If this type of light unit is provided, the possibility of its contact with fuel fumes should be considered. Installations that have this problem normally require the use of light units qualified as explosion proof. The installation should also be reviewed to determine if a single failure can cause the light to be on in the stowed position. If the light can be on, the potential for overheating or fire in the adjacent area should be considered.

AC 29.1385. § 29.1385 POSITION LIGHT SYSTEM INSTALLATION. Refer to AC 20-74, Aircraft Position and Anticollision Light Measurements, July 29, 1971.

AC 29.1387. § 29.1387 (Amendment 29-9) POSITION LIGHT SYSTEM DIHEDRAL ANGLES. Refer to AC 20-74.

AC 29.1389. § 29.1389 POSITION LIGHT DISTRIBUTION AND INTENSITIES. Refer to AC 20-74.

AC 29.1391. § 29.1391 MINIMUM INTENSITIES IN THE HORIZONTAL PLANE OF FORWARD AND REAR POSITION LIGHTS. Refer to AC 20-74.

AC 29.1393. § 29.1393 MINIMUM INTENSITIES IN ANY VERTICAL PLANE OF FORWARD AND REAR POSITION LIGHTS. Refer to AC 20-74.

AC 29.1395. § 29.1395 MAXIMUM INTENSITIES IN OVERLAPPING BEAMS OF FORWARD AND REAR POSITION LIGHTS. Refer to AC 20-74.

AC 29.1397. § 29.1397 (Amendment 29-7) COLOR SPECIFICATIONS. Refer to AC 20-74.

AC 29.1399. <u>§ 29.1399 RIDING LIGHT</u>.

a. Explanation. The riding light is an amphibious operation requirement. The function of this light is to make the rotorcraft visible at night to other vessels when the rotorcraft has landed on water. A very important point which should be remembered is that when a rotorcraft has landed on the water and is not in flight, it is considered a vessel in accordance with the United States Coast Guard (USCG) navigation rules (Inland Navigation Rules Act of 1980). If water operations are contemplated, one should acquire the USCG Navigation Rules, COMDTINST M16672.2A, which are for sale from Superintendent of Documents, U.S. Government Printing Office, Washington, D.C. 20402.

b. Procedures. A white light should be installed in a position where it will show the maximum unbroken light for a horizontal arc of 360° around the rotorcraft. If possible, this light should not be obscured by sectors of more than 6°. The light should be installed to meet the malfunction requirements of § 29.1309(b) (reference paragraph AC 29.1309). For the purpose of this light, the following definition found in the Inland Navigation Rules, 33 CFR 84.13, Color specification of lights, and 33 CFR 84.15, Intensity of lights, applies:

(1) The chromaticity of white lights shall conform to the following standards, which lie within the boundaries of the area of the diagram specified for each color by the International Commission on Illumination (CIE), in the "Colors of Light Signals," which is incorporated by reference. It is Publication CIE No. 2.2 (TC-1.6), 1975, and is available from the Illumination Engineering Society, 345 East 47th Street, New York, NY 10017. It is also available for inspection at the Office of the Federal Register, Room 8401, 1100 L Street NW., Washington, D.C. 20408.

(2) The boundaries of the area for white are given by indicating the corner coordinates, which are as follows:

X	0.525	0.525	0.452	0.310	0.310	0.443
Y	0.382	0.440	0.440	0.348	0.283	0.382

and 33 CFR 84.15 defines the required luminosity to be visible on a clear night for 2 nautical miles. The minimum luminosity of the light is given by the formula:

$$I = 3.43 \times 10^6 \times T \times D^2 \times K^{-D}$$

where: I is luminous intensity in candelas under service conditions,
T is threshold factor 2×10^{-7} lux,
D is range of visibility (luminous range) of the light in nautical miles, and
K is atmospheric transmissivity. For the prescribed lights the value of K shall be 0.8, corresponding to a meteorological visibility of approximately 13 nautical miles.

(3) Solving this formula indicates a minimum intensity of 4.3 candelas is required for this light.

NOTE: The FAR and the USCG navigation rules may be satisfied by an externally hung light(s). One method of compliance would be to use USCG approved all-around lights which are of the appropriate luminosity and externally hung.

AC 29.1401. <u>§ 29.1401 (Amendment 29-11) ANTICOLLISION LIGHT SYSTEM</u>.

(1) Certification for night operations requires an approved aviation red anticollision light. Determination of the location and how many anticollision lights are required to satisfy the regulations are functions of aircraft shape and the ability to obtain the required area coverage and light intensity. A detailed explanation of how to calculate the measured area coverage required by § 29.1401(b) is given in AC 20-30B. An explanation of the methods used to measure and calculate the light intensity and color required by § 29.1401(e) are explained in AC 20-74.

(2) The anticollision light(s) should be located to obtain the required coverage and to prevent cockpit reflections that would affect the crew's vision. The anticollision lights are required to be red to reduce cockpit reflections and objectionable effect of rotor blade strobing. During the period of August 11, 1971, through February 4, 1976, white lights were permitted by the rules; however, white lights resulted in undesirable cockpit reflections at night and in close proximity to clouds. For these reasons, white lights are not considered to be satisfactory in all operating conditions.
Section 29.1401(b) was changed in 1976 to require a red anticollision light. White lights have been approved for installation on rotorcraft when they were installed in addition to the required red lights, if an independent control for the white light was provided that allowed the pilot to eliminate any adverse cockpit reflections.

SUBPART F - EQUIPMENT

SAFETY EQUIPMENT

AC 29.1411. § 29.1411 SAFETY EQUIPMENT - GENERAL.

a. Explanation.

(1) This section contains requirements for the accessibility and stowage of required safety equipment. Compliance with this section should assure that:

(i) Locations for stowage of all required safety equipment have been provided.

(ii) Safety equipment is readily accessible to both crewmembers and passengers, as appropriate, during any reasonably probable emergency situation.

(iii) Stowage locations for all required safety equipment will adequately protect such equipment from inadvertent damage during normal operations.

(iv) Safety equipment stowage provisions will protect the equipment from damage during emergency landings when subjected to the inertia loads specified in § 29.561.

(2) It is a frequent practice for the rotorcraft manufacturer to provide the substantiation for only those portions of the ditching requirements relating to aircraft flotation and ditching emergency exits. Completion of the ditching certification to include the safety equipment installation and stowage provisions is then left to the affected operator so that those aspects can best be adopted to the selected cabin interior. In such cases, the "Limitations" section of the Rotorcraft Flight Manual should identify the substantiations yet to be accomplished in order to justify the full ditching approval. The operator (or modifier) performing these final installations is then concerned directly with the details of this paragraph. Any aspects of the basic rotorcraft flotation and emergency exits approval that are not compatible with the modifier's proposed safety equipment provisions should be resolved between the type certificate holder and the modifier prior to FAA/AUTHORITY approval for ditching.
(See paragraphs AC 29.801a(9) and AC 29.1415a(3).)

b. Procedures.

(1) A cockpit evaluation should be conducted to demonstrate that all required emergency safety equipment to be used by the crew will be readily accessible during any probable emergency situation. This evaluation should include, for example, emergency flotation equipment actuation devices, remote life raft releases, hand fire extinguishers, and protective breathing equipment.

(2) Stowage provisions for safety equipment shown to be compatible with the vehicle configuration presented for certification should be provided and identified so that:

(i) Equipment is readily accessible regardless of operational configuration.

(ii) Stored equipment is free from inadvertent damage from passengers and handling.

(iii) Stored equipment is adequately restrained to withstand the inertia forces specified in § 29.561(b)(3) without sustaining damage.

(3) For rotorcraft required to have an emergency descent slide or rope according to § 29.809(f), the stowage provisions for these devices must be located at the exits where they are intended to be used.

(4) Life raft stowage provisions should be sufficient to accommodate rafts for the maximum number of occupants for which certification for ditching is requested.

(i) Life rafts stowed inside the rotorcraft should be located near the ditching emergency exits so that:

(A) Life rafts are readily accessible and deployment through ditching emergency exits by passengers and crew may be accomplished without unreasonable effort and training.

(B) Deployment of life rafts can be accomplished without damage (i.e., punctures, tears, etc.).

(ii) Life rafts stowed outside of the rotorcraft should have--

(A) A readily accessible deployment device; and

(B) A secondary method of deployment near the stowed area.

(iii) Rotorcraft fuselage attachments for the life raft static lines required by § 29.1415(b)(2) must be provided.

(A) Static line fuselage attachments should not be susceptible to damage when the rotorcraft is subjected to the maximum emergency ditching water entry loads established by § 29.801. (See paragraph AC 29.801b(1).)

(B) Static line fuselage attachments should be structurally adequate to restrain a fully loaded raft of the maximum capacity required for ditching certification.

(C) Life rafts that are remotely or automatically deployed must be attached to the rotorcraft by the required static line after deployment without further action from the crew or passengers.

(5) Stowage provisions for the emergency locator transmitter (ELT) required by § 29.1415 must be located near a designated ditching emergency exit. The TSO under which most life rafts are approved and the operating regulations (e.g., 135.167(b)) require that the ELT be actually attached to an approved life raft. Configurations supplying an ELT as a part of an approved life raft package have been accepted as meeting the intent of § 29.1411(e).

(6) If stowage provisions for life preservers are included in an interior configuration, each life preserver when stowed must be within easy reach of each occupant while seated.

AC 29.1413. <u>§ 29.1413 (Amendment 29-16) SAFETY BELTS: PASSENGER WARNING DEVICE</u>.

a. <u>Explanation</u>. A safety belt design feature and a design feature for the belt warning or signal device are stated in the standard.

(1) Belts must have metal-to-metal latches (Amendment 29-16). Section 29.785(c), (f), and (g) of Amendment 29-24 concern design and installation standards for belts.

(2) Whenever a “fasten” seat belt sign or equivalent symbol is used, each pilot shall be able to control or operate the sign.

(3) Section 29.853(c) of Amendment 29-18 concerns illuminated “no smoking” information signs which are typically adjacent to any seat belt information sign. Whenever the crew and passenger compartments are separated, illuminated signs are required. However, a placard may be used to prohibit any smoking.

(4) TSO-C22, Safety Belts, contains acceptable aircraft belt standards. Also, TSO-C114, Torso Restraint Systems, dated March 27, 1987, contains acceptable aircraft standards, provided there is compliance with § 29.785.

b. <u>Procedures</u>.

(1) A TSO-C22 or TSO-C114 approved seat belt or seat belt/harness should be used. The rated load shall not be exceeded. During an interior compliance inspection, the belt shall be checked for proper label, rating, and metal-to-metal latches. Other features are required by § 29.785(c) and (g) of Amendment 29-24.

(2) A placard, legible to each passenger seated in the cabin, stating “fasten seat belts” (and harness if appropriate) may be used. This is similar to the “no smoking” placard standard.

(3) If an illuminated “fasten seat belt” sign or symbol is used, it should be legible to each seated passenger and must be controllable from each pilot seat.

AC 29.1415. § 29.1415 (Amendment 29-30) DITCHING EQUIPMENT.

a. Explanation.

(1) Emergency flotation and signaling equipment is not required for all rotorcraft overwater operations. However, if such equipment is required by an operating rule (e.g., § 135.167), the equipment supplied for compliance with the operating rule must meet the requirements of this section.

(2) Compliance with the provisions of § 29.801 for rotorcraft ditching requires compliance with the safety equipment stowage requirements and ditching equipment requirements of §§ 29.1411 and 29.1415, respectively.

(i) Emergency flotation and signaling equipment installed to complete certification for ditching or required by any operating rule must be compatible with the basic rotorcraft configuration presented for ditching certification. It is satisfactory if operating equipment is not incorporated at the time of original type certification of the rotorcraft provided suitable information is included in the “Limitations” section of the Rotorcraft Flight Manual to identify the extent of ditching certification not yet completed.

(ii) When the ditching equipment required by § 29.1415 is being installed by a person other than the applicant who provided the rotorcraft flotation system and ditching emergency exits, special care must be taken to avoid degrading the functioning of the aircraft devices and to make the ditching equipment compatible with them. (See paragraphs AC 29.801a(9) and AC 29.1411a(2).)

b. Procedures.

(1) Life rafts and life preservers used to show compliance with the ditching requirements must be of an approved type. Compliance with the requirements of TSO-C12 for life rafts and TSO-C13 for life preservers will satisfy regulatory requirements for approval of this equipment.

(i) Life preservers.

(A) Life preservers should comply with the requirements of the applicable operating regulations (FAR Parts 91, 135, 121, etc.). For extended overwater operations each life preserver is required to have an approved survivor locator light by the operating rules.

(B) Protective covers for life preservers should be compatible with the TSO requirements under which the basic life preserver was approved.

(ii) Life rafts.

(A) Life rafts are rated during their approval to the number of people that can be carried under normal conditions and the number that can be accommodated in an overload condition. Only the normal rating may be used in relationship to the number of occupants permitted to fly in the rotorcraft.

(B) The life raft configuration (i.e., number of life rafts and capacity of each raft) presented for ditching certification must be adequate to accommodate all rotorcraft occupants using the overload rating of the remaining raft(s) after the loss of one raft of the largest rated capacity. Thus, at least two rafts are required for any transport category rotorcraft extended overwater operation.

(C) Each life raft must be equipped with both a trailing line and a static line to be used for securing the life raft close to the rotorcraft for occupant egress. The static line should be of adequate strength to restrain the life raft under any reasonably probable sea state condition but must be designed to release before submerging the empty raft to which it is attached if the rotorcraft sinks.

(iii) Survival Equipment. Approved survival equipment if required by any operating rule must be attached to each life raft. Provisions for the attachment and stowage of the appropriate survival equipment should be addressed during the ditching equipment segment of the basic ditching certification.

(2) One emergency locator transmitter (ELT) meeting the applicable requirements of TSO-C91 must be provided for use in one life raft. The ELT provided for this purpose should be attached to one of the rafts or included in the survival equipment which is attached to one of the life rafts. If not attached to a life raft, the ELT must be located near an emergency ditching exit for compliance with § 29.1411(e). (See paragraph AC 29.1411b(5).)

AC 29.1419. § 29.1419 (Amendment 29-21) ICE PROTECTION.

NOTE: Section 29.877 was removed and replaced by § 29.1419 in Amendment 29-21. Guidance material for Ice Protection prior to Amendment 29-21 is retained in AC paragraph 29.877 (Subpart D).

a. Background.

(1) In March 1984, the FAA/AUTHORITY for the first time certificated a rotorcraft for flight into known icing conditions. Several other manufacturers are pursuing designs for icing flight capability.

(2) Most rotorcraft icing technology has been developed for military rotorcraft. The only U.S. military rotorcraft equipped and approved for flight into icing conditions is the UH-60A (Blackhawk). The UH-60A is limited to supercooled cloud conditions where liquid water content (LWC) does not exceed 1.0°gm/m^3 and outside air temperature (OAT) is not below -20° C.

(3) Many rotorcraft operators have voiced a high priority on obtaining rotorcraft approved for operation in icing conditions.

(4) The icing characteristics envelope of FAR Part 25, Appendix C, has served as a satisfactory design criteria for fixed-wing operations for two decades. The envelope, as presented, extends to 22,000 feet with possible extension to 30,000 feet but does not present icing severity as a function of altitude. At the time the envelope was derived, it was assumed that all transport category airplanes would operate to at least 22,000 feet. For present state-of-the-art rotorcraft, this assumption is not valid. As such, an altitude-limited icing envelope based on the same data used to derive the Part 25, Appendix C, and the Part 29, Appendix C, envelopes is presented as an alternate to the full icing envelope.

b. Explanation.

(1) General.

(i) The discussion in this paragraph pertains generally to certifications to the full icing envelope of Part 29, Appendix C, within the altitude limitations of the rotorcraft or to the altitude-limited icing envelope based on a 10,000-foot pressure altitude limit. The actual icing envelope considered may be further restricted based on the actual pressure altitude envelope for which certification is requested. It envisions certification with full ice protection systems (rotor blades, windshields, engine inlets, stabilizer surfaces, etc.). With the exception of pilot controllable variables such as altitude and airspeed, limited certification (either in terms of icing envelope or protection capability) is not envisaged at this time due to the difficulty in forecasting the severity of icing conditions, relating the effects of the forecasted conditions to the type of aircraft, and the effects of reported icing among various types of aircraft, particularly between fixed- and rotary-wing aircraft. In addition, with a limited protection capability, viable escape options may not be operationally available if limitations are exceeded.

(ii) The discussion in this paragraph, regarding rotor blade ice protection, is oriented primarily toward electrothermal rotor deicing systems, since these have the most widespread acceptance and projected use within the industry. Also, most of the testing and research into rotorcraft ice protection to date has been conducted with these types of systems. Research is continuing with other types of systems such as anti-icing fluid systems, and information will be added to address certification of these as necessary. It should also be noted that most of the rotorcraft icing experience accumulated to date has been on rotorcraft with symmetrical airfoil sections. The application of this experience to rotorcraft with asymmetrical airfoils should be carefully evaluated. Limited experience has been gained during development and qualification testing of the Army Blackhawk on asymmetrical airfoil icing characteristics. The most prominent difference appears to be a more rapid degradation of airfoil performance. Rapidity of performance degradation is also dependent upon severity of the icing condition (primarily a function of liquid water content) and ice shape (primarily a function of OAT and median volumetric droplet diameter (MVD)).

(iii) The effects of ice can vary considerably from rotorcraft to rotorcraft. Experience gained for a rotor system with an identical blade profile could provide valuable information but should be used cautiously when applied to another rotorcraft. Assumptions cannot necessarily be made based on icing test results from another rotorcraft. Particular care should be exercised when drawing from fixed-wing icing experience as the widely different and varying conditions seen by the rotor blades make many comparisons with fixed-wing results invalid. Likewise, icing effects on rotor blades vary significantly from those on other parts of the rotorcraft. This is due to changing blade velocity as compared with the constant velocity of the remaining parts.

(2) Reference Material. Prior to commencement of efforts to design and certify a rotorcraft, the references listed in paragraph d should be reviewed. FAA Technical Report ADS-4, Engineering Summary of Airframe Icing Technical Data, December 1963, although somewhat dated, is recommended for basic aircraft icing protection system design information.

(3) Objective. The objective of icing certification is to verify that throughout the approved envelope, the rotorcraft can operate safely in icing conditions expected to be encountered in service (i.e., Appendix C of Part 29 or the altitude-limited icing envelope presented herein). This will entail determining that no icing limitations exist or defining what the limitations are, as well as establishing the adequacy of the ice warning means (or system) and the ice protection system. A limiting condition may manifest itself in one of several areas such as handling qualities, performance, autorotation, asymmetric shedding from the rotors, visibility through the windshield, etc. Prior to flight tests in icing conditions, sufficient analyses should have been conducted to determine the design points for the particular item of the rotorcraft being analyzed (windshield, engine inlet, rotor blades, etc.). After the analyses are reviewed and found adequate, tests should be conducted to confirm that the analyses are valid and that the rotorcraft can operate safely in any supercooled cloud icing condition defined by Part 29, Appendix C, or the altitude-limited icing envelope. References d(1) and (3) may be useful in determining the design points and extrapolation of test data to the desired design points.

(4) Planning. For best utilization of both the applicant's and the FAA/AUTHORITY's resources, the applicant should submit a certification plan at the start of the design and development effort. The certification plan should describe all efforts intended to lead to certification and should include the following basic information:

Rotorcraft and systems description.
Ice protection systems description.
Certification checklist.
Description of analyses or tests planned to demonstrate compliance.
Projected schedules of design, analyses, testing, and reporting efforts.
Methods of test - artificial vs. natural.
Methods of control of variables.
Data acquisition instrumentation.
Data reduction procedures.

(5) Environment.

(i) Definitions.

(A) Supercooled Clouds. Clouds containing water droplets (below 32° F) that have remained in the liquid state. Supercooled water droplets will freeze upon impact with another object. Water droplets have been observed in the liquid state at ambient temperatures as low as -60° F. The rate of ice accretion on an aircraft component is dependent upon many factors such as droplet size, cloud liquid water content, ambient temperature, and component size, shape, and velocity.

(B) Ice Crystal Clouds. Glaciated clouds existing usually at very cold temperatures where moisture has frozen to the solid or crystal state.

(C) Mixed Conditions. Partially glaciated clouds at ambient temperatures below 32° F containing a mixture of ice crystals and supercooled water droplets.

(D) Freezing Rain and Freezing Drizzle. Precipitation existing within clouds or below clouds at ambient temperatures below 32° F where rain droplets remain in the supercooled liquid state.

(E) Sleet. Precipitation of transparent or translucent pellets of ice which have a diameter of 5mm or less.

(F) Hail. Solid precipitation in the form of balls or pieces of ice (hailstones) with diameters ranging from 5mm to more than 50mm.

(ii) Appendix C of Part 29 defines the supercooled cloud environment necessary for certification of rotorcraft in icing except that the pressure altitude limitation is that of the rotorcraft or that selected by the applicant, provided the remaining altitude envelope is operationally practical. Due to air traffic system compatibility constraints, approval of a maximum altitude less than 10,000 feet pressure altitude should be discouraged. However, there are operations where a lower maximum altitude has no effect on the air traffic system and would still be operationally useful. Figures 3 and 6 of Appendix C, Part 29, relate the variation of average LWC as a function of cloud horizontal extent. These relationships should be used for design assessment of the most critical combinations of conditions as a function of en route distance. This, in combination with a capability to hold in icing conditions for 30 minutes at the destination, is commensurate with policies previously established for fixed-wing aircraft. Figures 3 and 6 should be used in conjunction with the altitude-limited criteria of figures AC 29.1419-1 through -4 herein. It is emphasized that LWC extremes expressed in Part 29 Appendix C, criteria represent the maximum average values to be anticipated within an exceedance probability of 99.9 percent. Transient, instantaneous peak values of much higher LWC have been observed. These instantaneous peak values appear to be of little significance to the design of protected and unprotected surfaces; however, these high values, if encountered, may induce shedding of ice from some unprotected surfaces. This is due to radical changes in the rate of release of latent heat and resultant changes in the structural properties and adhesion force of ice.

(iii) An analysis performed at the FAA Technical Center in 1985 concludes that the aircraft icing environment below 10,000 feet is not as severe in terms of LWC and OAT as that depicted in the Part 29, Appendix C, envelope. This AC presents the altitude-limited envelope that may be employed by those applicants who elect to certify with a 10,000-foot pressure altitude limit. The altitude-limited envelope is based upon the same data that were used to derive the design criteria of Part 29, Appendix C (figures AC 29.1419-1 through -4). The data used to derive these limited envelopes cannot be used to further define icing conditions between 10,000 feet and 22,000 feet; hence, above 10,000 feet, the Part 29, Appendix C, envelopes should be used. It should be noted that the engine inlets should still meet the icing requirements of § 29.1093. The limited icing envelopes may be used on an equivalent safety basis to show compliance with the intent of § 29.1093 if the altitude limit established for the rotorcraft is not greater than 10,000 feet.

(iv) Significantly different effects can result from various combinations of parameters. For example, most rapid ice accumulations occur at the high values of liquid water content, although the greatest impingement area occurs at the high values of droplet size. Most critical ice shapes are a function of each of these parameters in addition to airspeed, surface temperature, and surface contour. Care should be taken to explore the entire specified ranges of these parameters during the design, development, and certification efforts.

(v) Mixed conditions (i.e., a combination of ice crystals and supercooled water droplets) and freezing rain or freezing drizzle are not addressed in the Part 29 environmental criteria but can present more severe icing conditions than those defined. Although the probability of encountering freezing rain is relatively low, mixed conditions commonly occur in supercooled cloud formations. Little data have been gathered on the effects of encountering mixed conditions (see paragraph AC 29.1419d(6)). There are no criteria for certification in mixed conditions or freezing rain at present. In addition to the hazards of operating any aircraft in icing, certain aspects of rotorcraft icing (relatively low altitude operation, asymmetric shedding with resulting vibration, and ice damage or ingestion) warrant a caution notice in the RFM advising that the rotorcraft is not certified for operation in freezing rain or freezing drizzle. Avoidance procedures (e.g., climb or descent) may also be useful.

(6) Flight Test Prerequisites.

(i) The prototype rotorcraft should be capable of IFR and IMC flight.

(ii) Sufficient analyses should be developed, submitted, and accepted by FAA/AUTHORITY to show that the rotorcraft is capable of safely operating to the selected design points of both the continuous maximum and intermittent maximum conditions of Part 29, Appendix C, or the altitude-limited icing envelope. A detailed failure modes and effects analysis (FMEA) of the ice protection system should be performed.

(iii) Specific attention should be given to (1) assuring that the selected design condition(s) of atmospheric and rotorcraft flight envelopes have been identified; (2) qualification and design of ice protection systems and components; and (3) component installation and ice formation effects upon basic rotorcraft structural properties and handling qualities. These assurances can be established from analyses, bench tests, and/or dry air flight tests or simulated icing tests, as appropriate, prior to flight tests in natural icing.

(iv) The applicant should assess rotor blade stability with ice deposits to assure that dynamic instability will not occur in icing conditions. This assessment may be accomplished by analysis including consideration of failure of the most critical segment of the rotor blade ice protection system. It also may be accomplished by experimental means such as attaching dummy ice shapes to the blades and using a whirl stand or wind tunnel.

c. Procedures.

(1) Compliance.

(i) In general, compliance can be established when there is reasonable assurance that while operating in the specified icing environment (1) the engine(s) will not flameout or experience significant power losses or damage; (2) stress levels are not reached with ice accumulations that can endanger the rotorcraft or cause serious reductions in component life; (3) the handling qualities, performance, visibility, and systems operation are defined and are not deteriorated unacceptably; (4) inlet, vent, or drain blockage (such as fuel vent, engine, or transmission cooler) is not excessive; and (5) autorotation characteristics are acceptable with maximum ice accretion between de-ice cycles. Assessment of performance loss should include not only the drag and weight of the ice itself but electrical or other load demands of the ice protection system and any performance changes resulting from modified rotor blade contours.

(ii) It is emphasized that ice formations (shape, weight, etc.) vary significantly under varying conditions of OAT, LWC, MVD, airspeed, attitude, and rotor RPM. The most critical conditions should be defined by means of analyses or test and verified by test. Performance changes under these various conditions should be determined and found acceptable.

(iii) Laboratory, icing tunnel, ground spray rig, and airborne icing tanker tests are all very useful in developing an ice protection capability, but none of these, either individually or collectively, can satisfy the full requirements for certification. None can presently duplicate the combinations of liquid water content, droplet size, flow field, and random shedding patterns found in natural icing conditions. Airborne tankers hold considerable promise of being able to fulfill certification requirements (in addition to the advantage of being able to produce an icing environment on demand rather than having to wait for it to occur in nature), but tankers have not been able to generate droplet sizes that cover the complete envelope for certification. Many improvements have been made in some tankers in recent years; however, large droplet sizes have typically been a problem. Also, the size of existing tanker clouds is not of sufficient cross section to immerse the entire rotorcraft. There are also solar radiation and relative humidity effects to be considered and correlated with natural icing when using a tanker. The tanker should be able to immerse the entire rotor system as a minimum and should have a means of controlling and changing the cloud characteristics uniformly and repeatably. Until an artificial method has been successfully demonstrated and accepted, icing certification should include flight tests in natural icing conditions.

(iv) Flight testing in natural icing conditions also has limitations. Paragraph AC 29.1419d(16) contains information that may be useful in planning natural icing flight tests. The key limitation of natural icing flight tests is being able to find the combinations of conditions that comprise critical design points. This is especially true of those points falling near the 99.9 percentile of exceedance probability; e.g., high LWC at low OAT with large MVD. It is emphasized that some more severe design points, however, may exist within the atmospheric icing envelope rather than near the edges or corners of the envelope. This does not mean that natural icing tests must be conducted at all the selected design conditions. Natural icing tests should be conducted in conditions as close to design points as possible and sufficient correlation shown with the analyses to assure that the rotorcraft can operate safely throughout the design envelope.

(v) Certification flight testing should be extensive enough to provide reasonable assurance that either induced or random ice shedding does not present a problem. The most likely indication of a problem if it exists will be ice impact on the airframe or rotor imbalance resulting in vibration. The following should be considered sufficient for rejection:

(A) Vibrations sufficient to make the instruments difficult to read accurately.

(B) Vibrations sufficient to exceed the structural or fatigue limits of any rotorcraft part such as blade, mast, or transmission components.

(C) Ice impact damage to essential parts, such as the tail rotor, that could create a flight hazard. Cosmetic, nonstructure flaws that do not exceed wear and tear characteristics or maintenance criteria are acceptable. Any ice shedding effects that require immediate maintenance action are unacceptable.

(vi) There should be a means identified or provided for determining the formation of ice on critical parts of the rotorcraft which can be met by a reliable and safe natural warning or an ice detection system. A system utilizing OAT must include an accurate OAT measurement since the onset of icing can occur in a very narrow temperature band requiring sensitive and accurate OAT measurement. OAT accuracy should be relative to the true temperature of the air mass. Total system accuracy should be ±0.5° C in the -5.0° to +5.0° C range and ±1° C throughout the remaining temperature range. The location of the sensor has been shown to be very critical and, in effect, there can be a position error or other errors induced by ice formations or solar radiation. If the system measures liquid water content, consideration should be given to the fact that the actual LWC fluctuates considerably as the rotorcraft passes through an icing environment. A warning system displaying or utilizing a peak or average LWC value (rather than an instantaneous readout) should include sufficient conservatism to provide a margin of safety. The value of an LWC detecting system lies in its utility as a warning that ice is being encountered. The actual magnitude of LWC in combination with OAT and MVD can be used to indicate the icing severity level. The U.S. Army is currently developing an advanced ice detection system for potential application to rotorcraft.

(2) Instrumentation and Data Collection.

(i) Instrumentation proposed for certification tests, including flight strain surveys, should be reviewed as early as possible in the program to establish that it will provide the necessary data. The need for accurate OAT measurement previously noted for operation in icing also applies to the certificated configuration. Mechanical devices such as the rotating multicylinder and rotating disc have been used for measuring the ice accretion rate which is related by calibration to average LWC and MVD. More recently, hybrid mechanical/electronic LWC measuring devices have been used. Devices that rely on ice accretion as a signal source are subject to the Ludlam limit (the limits whereby latent heat of fusion is not totally absorbed, thus resulting in incomplete freezing of the moisture and some inaccuracy in the indication). The Ludlam limit is a function of various parameters including OAT, airspeed, LWC, and MVD. The Ludlam limit may vary from one device to another. (See paragraphs AC 29.1419d(8) and d(9)(i) for further information). Gelatin slides, soot and oil slides, and more recently, laser nephelometers have been used to measure droplet size. Other calibrated devices intended for measurement of LWC should be used. Paragraph AC 29.1419d(16) describes several of these devices. Photographic coverage of critical areas may be necessary to ascertain that ice protection systems are functioning properly and that there are no runback problems. (The term “runback” refers to liquid water that has not been evaporated by surface de-ice equipment and flows back to an unheated area subject to freezing.) Paragraph AC 29.1419d(19) highlights use of video techniques and equipment for this purpose. Some systems will require acceptable calibration techniques and data.

(ii) Gelatin, soot, and oil slides provide data that can be used to estimate MVD at discrete intervals while laser nephelometer data can provide time histories of MVD droplet size distributions. Gelatin slide data should be taken frequently during test flights to properly characterize the cloud. Laser nephelometer data have been found to be highly dependent upon knowledge of the equipment and calibration. Proper calibration, maintenance, and data processing techniques should be utilized and demonstrated. Additional information on the subject may be found in paragraph AC 29.1419d(18).

(iii) Structural instrumentation requirements should also be established as early as possible in the program. Flight strain measurements are strongly recommended in assessing the ice imposed stress on the rotorcraft. The flight strain measurements should determine the effect on fatigue life due to ice accumulation for such items as main rotor blades, main rotor hub components, rotating and fixed controls, horizontal stabilizer, tail rotor, etc. The subsequent proper operation of retractable devices such as landing gear should be demonstrated with representative ice accretion. In addition, the static and fatigue strength of the blade with heater mat must be substantiated. Any effect of the heater mat on fatigue strength of the blades must be considered.

(3) Additional Considerations. The following are items to consider in an icing certification program. They are not intended to be all-inclusive, and the possibility of widely differing characteristics and critical areas among various rotorcraft in icing should be considered.

(i) The rotorcraft should be shown by analysis and confirmed by either simulated or natural icing tests to be capable of holding for 30 minutes in the design conditions of the continuous maximum icing envelope at the most critical weight, CG, and altitude with a fully functional ice protection system.

(ii) A single ice protection system and power source may be considered acceptable provided that after any single failure of the ice protection system, the rotorcraft can be shown by analysis and/or test to be capable of safe operation (no hazard) for 15 minutes following failure recognition in the continuous icing envelope used as the basis for certification within the same icing limits used for the 30-minute hold criteria. During this 15-minute period the rotorcraft may exhibit degraded characteristics. Pilot controllable operating limitations such as airspeed may be used to satisfy this continued safe flight criteria. For purposes of determining performance and handling qualities degradation, ice protection system failure need not be considered to occur simultaneously with engine failure unless ice protection system operation is dependent upon engine operation.

(iii) Although current airborne weather radar technology systems may be useful in avoiding potential icing conditions by detecting precipitation, the use of weather radar is not an FAA/AUTHORITY requirement for icing certification.

(iv) If the ice protection is not operating continuously, there must be a means to advise the crew when the rotorcraft is in icing conditions in order that the system may be activated.

(v) No autorotational performance data is required for rotorcraft which have Category A powerplant installations. All rotorcraft certified for flight in icing conditions must be capable of full autorotational landings with the ice protection system operating. Autorotational entry, steady state, and flare entry flying qualities and performance should be evaluated with an ice load. Since the Category A en route performance can vary as the ice protection system operates, a mean value of cyclic torque is acceptable provided, at no time, the rate of climb falls below zero. The rotorcraft is assumed to be clear prior to takeoff, and, therefore, the takeoff performance is not degraded. The landing performance can be based on the in-flight assessment of overall performance degradation. Items such as fuel burns can be used as part of the in-flight performance degradation determination. Regardless of the methods used to determine performance degradation, they must be easily used by the crew. The hover performance should be addressed for the termination of a flight after an icing encounter. The engines must be protected from the adverse effects of ice. When ice does accumulate on the inlets, screens, etc., it must be accounted for in performance, engine operating characteristics, and inlet distortion.

(vi) The handling qualities of the rotorcraft must be substantiated if ice can accumulate on any surface. When ice can accumulate on unprotected surfaces, the rotorcraft must exhibit satisfactory VFR/IFR handling qualities. In addition, following the failure of the de-ice system, the rotorcraft must be safely controllable for 15 minutes, i.e., the rotorcraft must be free from excessive and rapid divergence. Artificial ice shapes may be acceptable for acquisition of flight test data necessary for handling qualities and performance evaluations and demonstrations.

(vii) Items such as fuel tank vents, cooling vents, antennas, etc., must be substantiated for maximum icing effects.

(viii) The ice protection system should be sufficiently reliable to perform its intended function in accordance with the requirements of § 29.1309. These requirements may in some instances be met by the use of sound engineering judgment during design and compliance demonstrations. In many instances, use of good design practices, failure modes and effects analysis, and similarity analyses combined with good judgment will be adequate. In some instances the need for reliability analyses may be desirable. Additional information pertaining to reliability is contained in paragraph AC 29.1309 (§ 29.1309).

(ix) The subject of lightning must be addressed. The criteria applied on rotorcraft with ice protection systems are that "the rotorcraft must be protected in such a manner to minimize lightning risk." The general rules of § 29.1309(a), (b), and (c) are applicable to ensure adequate lightning protection.

(x) Ice protection of pitot-static sources, windshields, inlets, exposed control linkages, etc., must be considered.

(xi) The impact of ice protection system failure, complete and partial, and achieving adequate warning thereof must be assessed.

(xii) The impact of delayed application of ice protection systems should be assessed. Hazardous conditions should not be apparent. Any rotorcraft characteristic changes resulting should be covered in cautionary material in the rotorcraft flight manual.

(xiii) Possible droop stop malfunction with ice accumulation and its potential hazard to the rotorcraft, its occupants, and ground personnel must be assessed.

(xiv) Possible ice shedding hazards to ground personnel or equipment in proximity to turning rotors following flight in icing conditions should be given much consideration.

(4) Flight Manual. Areas of the flight manual which may require input are:

(i) Operating limitations including approved types of operation and prohibiting operation in freezing rain or freezing drizzle conditions. Avoidance procedures may also be useful.

(ii) Normal Operating Procedures. Information on the ice detection means or system and ice protection system and their capabilities.

(iii) Emergency Operating Procedures. Operating procedures containing essential information particularly with system failure.

(iv) Caution Notes. These caution notes should advise or address:

(A) Against inducing asymmetric shedding with rapid control inputs or rotor speed changes, except possibly as a last resort. Rotor speed changes appear to be more effective than control inputs in removing ice from the rotor blades of some rotorcraft.

(B) Loss in range, climb rate, and hover capability following prolonged operation in icing.

(C) The need for clean blade surfaces and use of approved cleaning solvents or ground deicing/anti-icing agents prior to start of rotors turning.

(D) Changes in autorotational characteristics resulting from formations.

(E) If the rotorcraft has been certificated for flight in supercooled clouds and falling and blowing snow, flight in other conditions such as freezing rain, freezing drizzle, sleet, hail, and combinations of these conditions with supercooled clouds should be avoided.

(F) The potential hazards to ground personnel, passengers deplaning, and equipment in proximity to turning rotors following flight in icing conditions.

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(ii) Artificial Icing Tests, UH-1H Helicopter, Part 1, Final Report, January 1974, USAASTA Project No. 73-04-4.

(iii) Artificial Icing Tests, UH-1H Helicopter, Part II, Heated Glass Windshield, Final Report, USAASTA Project No. 73-04-4.

(iv) Artificial Icing Tests, Lockheed Advanced Ice Protection System Installed on a UH-1H Helicopter, Final Report, June 1975, USAAEFA Project No. 74-13.

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(vi) Microphysical Properties of Artificial and Natural Clouds and Their Effects on UH-1H Helicopter Icing, Report USAAEFA Project No. 78-21-2.

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(viii) Artificial and Natural Icing Tests of the YCH-4TD, Final Report, May 1981, USAAEFA Project No. 79-07.

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(6) Further Icing Experiments on an Unheated Nonrotating Cylinder, National Research Council, Canada Report LTR-LT-105, dated November 1979, by J.R. Stallabrass and P.F. Hearty.

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(19) An Appraisal of the Single Rotating Cylinder Method of Liquid Water Content Measurement, National Research Council Canada Report LTR-LT-92, dated November 1978, by J.R. Stallabrass.

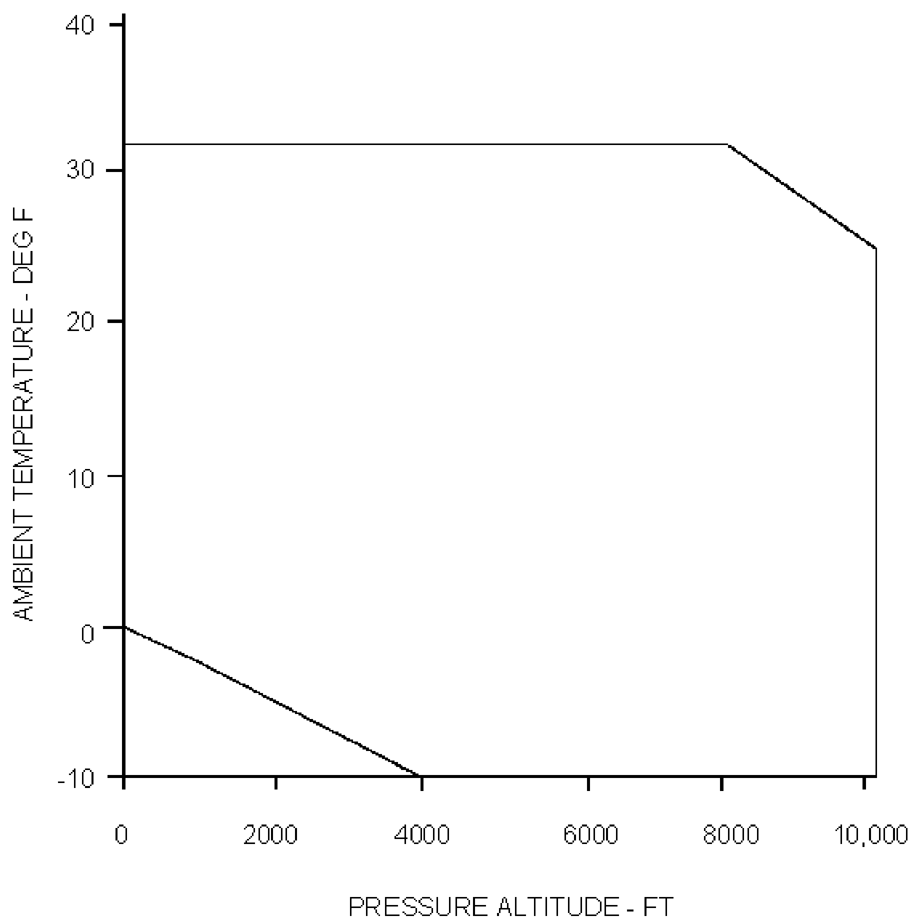

FIGURE AC 29.1419-1 CONTINUOUS ICING - TEMPERATURE VS ALTITUDE LIMITS

Figures AC 29.1419 -1 through 4 represent the approach to a 10,000-foot altitude limit. See Paragraph b(5)(iii) for a discussion of this approach.

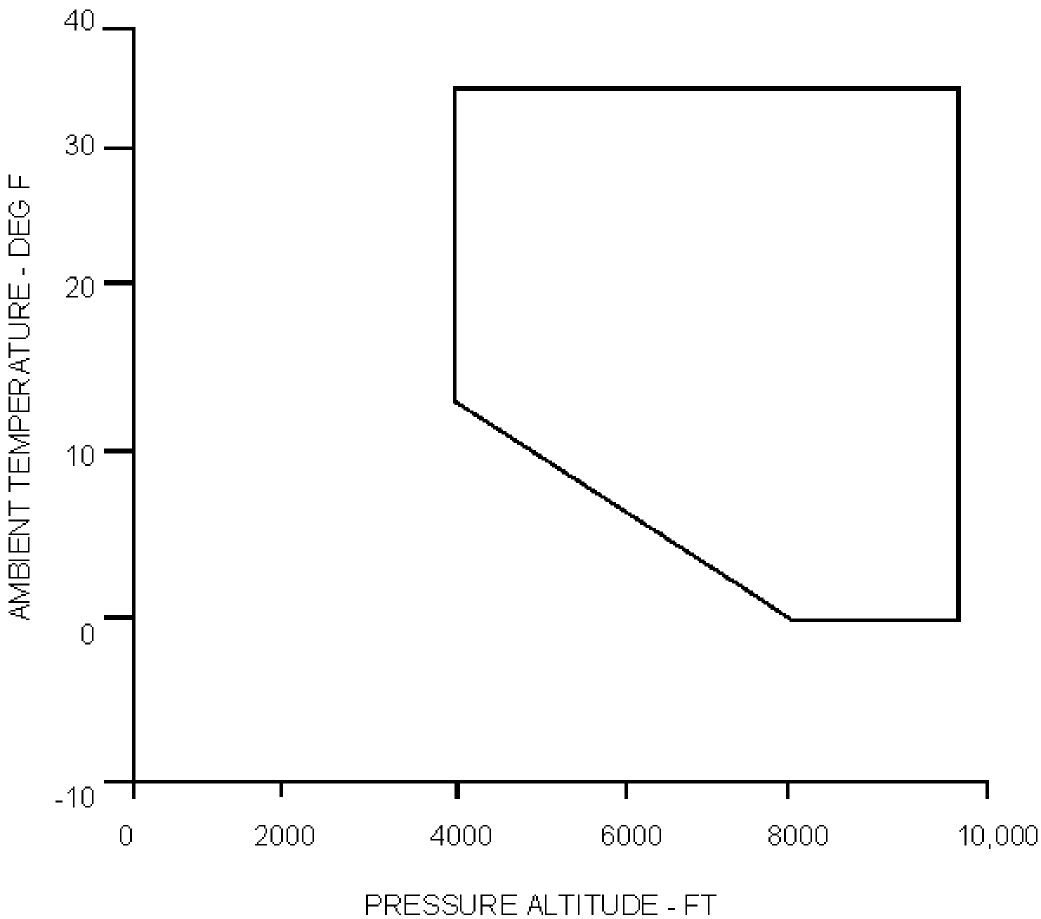

FIGURE AC 29.1419-2 INTERMITTENT ICING - TEMPERATURE VS ALTITUDE LIMITS

Figures AC 29.1419-1 through 4 represent the approach to a 10,000-foot altitude limit. See Paragraph b(5)(iii) for a discussion on this approach.

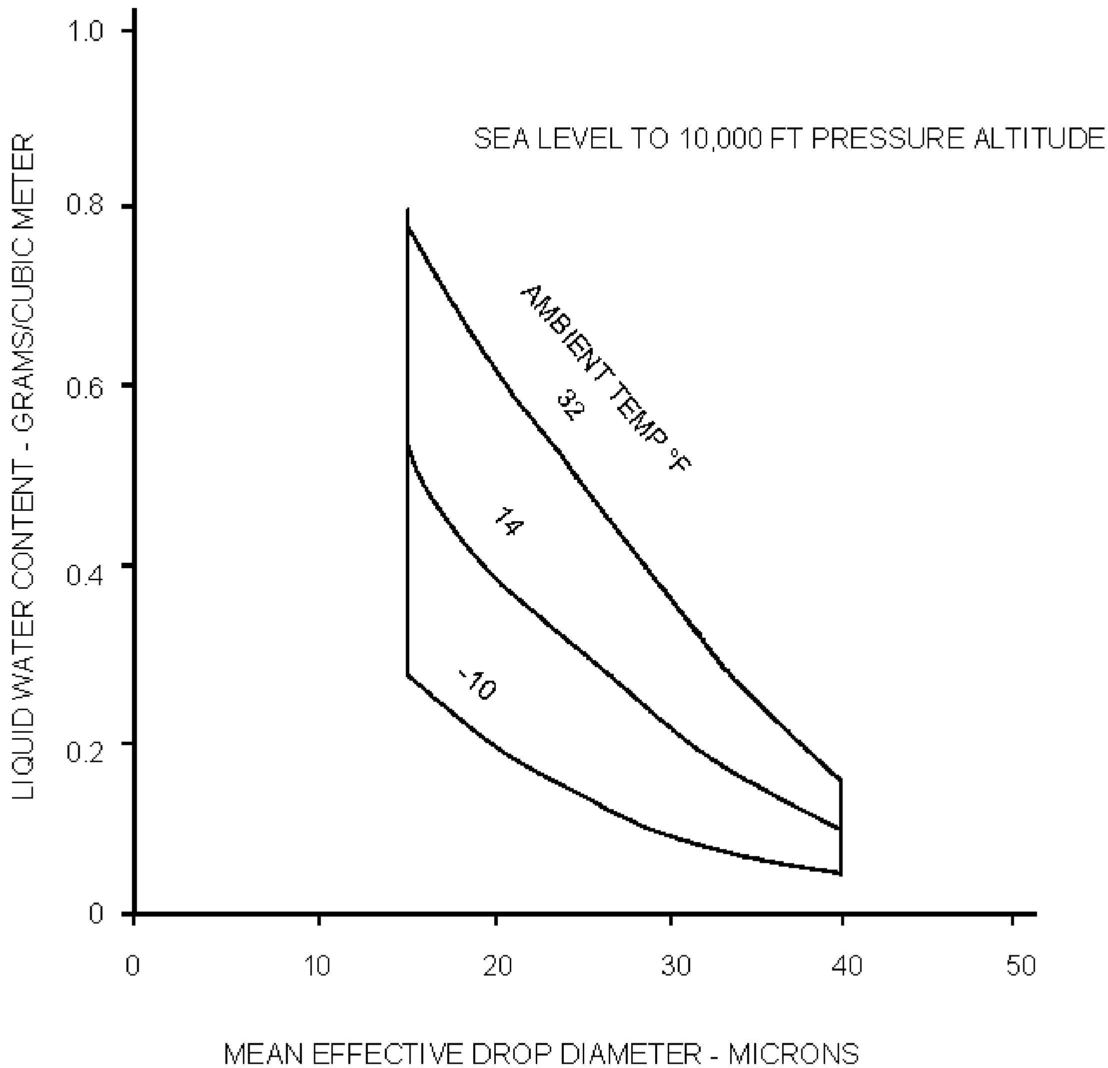

FIGURE AC 29.1419-3 CONTINUOUS ICING - LIQUID WATER CONTENT VS. DROP DIAMETER

Figures AC 29.1419-1 through 4 represent one approach to a 10,000-foot altitude limit. See Paragraph b(5)(iii) for a discussion on this approach.

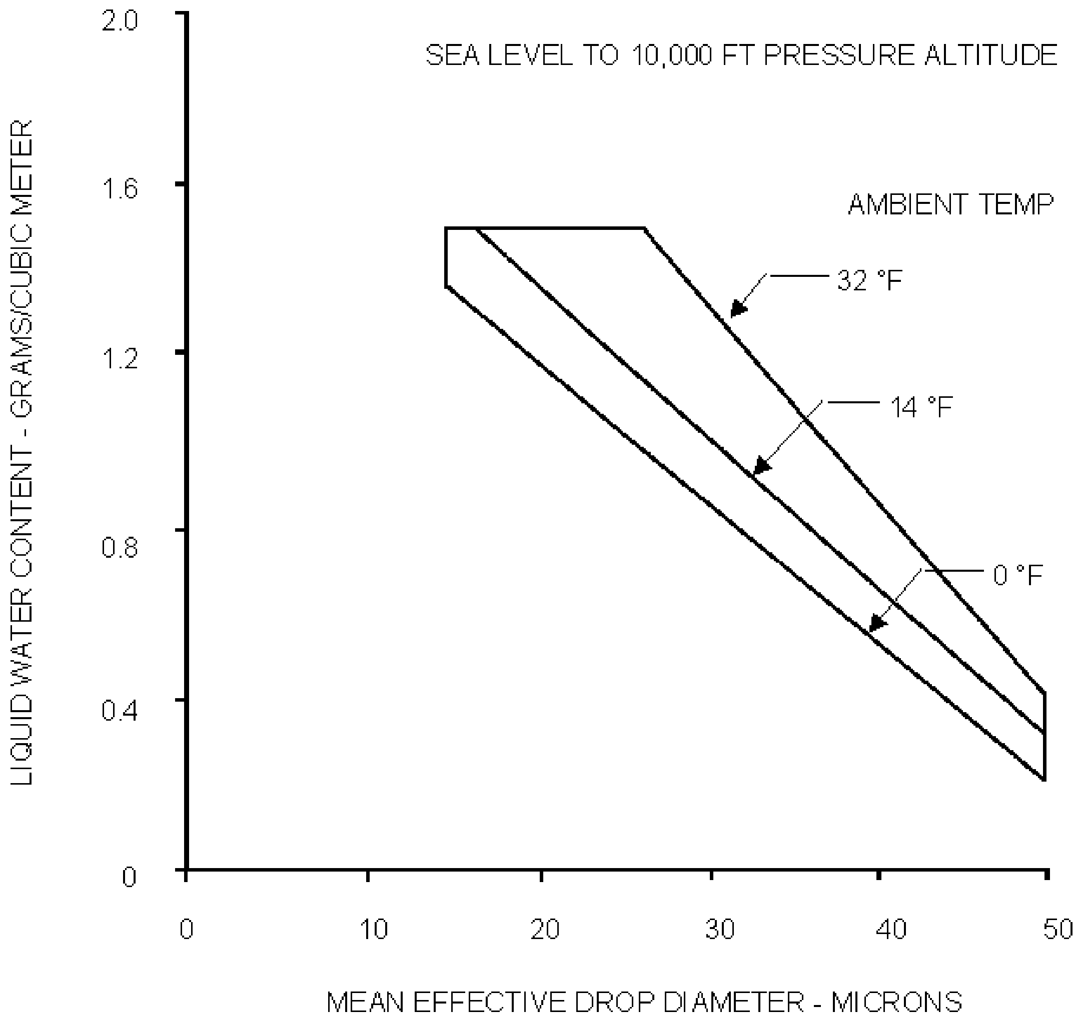

FIGURE AC 29.1419-4 INTERMITTENT ICING - LIQUID WATER CONTENT VS DROP DIAMETER

Figures AC 29.1419-1 through 4 represent one approach to a 10,000-foot altitude limit. See Paragraph b(5)(iii) for a discussion of this approach.

SUBPART F - EQUIPMENT

MISCELLANEOUS EQUIPMENT

AC 29.1431. § 29.1431 ELECTRONIC EQUIPMENT.

a. Background. This section contains some specific requirements for electronic equipment in the rotorcraft. The principal requirements of this section are that radio and navigation equipment must be free from hazards, both in themselves and in their effect on any other items installed in the rotorcraft, and that operation of the radio and navigation equipment does not interfere with operation of any other required avionics.

b. Procedures. In showing compliance with this section, tests and analysis should be performed as necessary to determine that:

(1) All radio and navigation equipment is installed and operated in such a manner that it does not result in hazards to the rotorcraft. It also should not have an effect on any other components of the rotorcraft such that a hazardous condition is created. Note that consideration should be given to the effects of critical environmental conditions.

(2) All radio and navigation systems and equipment should be installed and operated in a manner that will not have a detrimental effect on the proper functioning of any electronic equipment or system required by the FAR. It should be noted that §§ 29.1301 (reference paragraph AC 29.1301) and 29.1309(b) through (d) (reference paragraph AC 29.1309) apply to all installed equipment and systems and § 29.1309(a) applies to all systems and equipment required by Parts 21 through 49. As an example of showing compliance with this section, consider a high frequency radio (HF) system installation. The first thing to determine is that the installation and operation of the HF system cannot create a hazard. Consideration may be necessary in hazardous situations such as precipitation on the antenna. Next, it should be determined that the operation of the HF does not cause interference to a system whose functioning is required by the FAR. An example of unacceptable interference would be if operating the HF transmitter caused one of the navigation radios to malfunction.

(3) Finally, it should be determined that other systems do not interfere with the HF system. Additional guidance on the testing of avionics equipment and installation is contained in paragraph AC 29 MG 1.

AC 29.1433. § 29.1433 VACUUM SYSTEMS.

a. Explanation. Vacuum systems have been utilized on some rotorcraft to provide an energy source for the flight instruments. This specific rule addresses the potential hazards which are peculiar to vacuum system installations. The possible fire hazards presented by these systems are of particular concern.

b. Procedure. The following items should be specifically addressed when evaluating a vacuum system installation:

(1) Pressure and Temperature Protection. The high-pressure outlet of the vacuum pump should have a means to automatically relieve the pressure if it becomes excessively high or the air temperature becomes excessively hot.

(2) Fire Hazard Protection. The components of the vacuum system that are mounted in a designated fire zone should be fire resistant. This includes engine or transmission driven pumps if they are in a fire zone. The discharge side of the pump may emit flammable fluids. This discharge side of the pump, along with its associated lines and fittings, should meet the criteria in paragraph AC 29.1183.

AC 29.1435. § 29.1435 HYDRAULIC SYSTEMS.

a. Reference Regulations. The following sections of Part 29 are either incorporated in the provisions of § 29.1435 or are otherwise applicable to hydraulic system design:

(1) Section 29.695. Paragraph AC 29.695 covers power boost and power operated control systems.

(2) Section 29.861. Paragraph AC 29.861 covers fire protection of structure, controls, and other parts.

(3) Section 29.863. Paragraph AC 29.863 covers flammable fluid fire protection.

(4) Section 29.1183. Paragraph AC 29.1183 covers lines, fittings, and components.

(5) Section 29.1185. Paragraph AC 29.1185 covers flammable fluids.

(6) Section 29.1189. Paragraph AC 29.1189 covers shutoff means.

(7) Section 29.1309. Paragraph AC 29.1309 covers the requirements for functioning and reliability, and prevention of hazards if malfunctions or failures occur.

(8) Section 29.1322. Paragraph AC 29.1322 covers warning, caution, and advisory lights.

b. System Design. It is assumed that the hydraulic system is to be utilized to operate the primary control system of the rotorcraft and the rotorcraft cannot be safely operated without the hydraulic system.

(1) Section 29.1309, paragraphs (a) and (b), provides for functioning reliably under any foreseeable operating condition and prevention of hazards after any malfunction or failure.

(2) The substantiating data should include a failure analysis that considers every possible system component failure, such as (but not limited to) ruptured lines, pump failure, regulator failure, ruptured seals, clogged filters, broken pilot valve connections, etc.

(3) The requirements of § 29.1309(a) and (b) are met by dual independent hydraulic systems from the reservoir, hydraulic pump, regulator, connecting tubing, and hoses through the actuators. There must be no commonality in the fluid-containing components. A break in one system should not result in fluid loss in the remaining system.

(4) The pumps should be separated as far as practicable; i.e., on opposite sides of the rotor drive transmission, on separate engines, or one pump on an engine and the other on the rotor drive transmission. The tubing and hoses should also be routed with as much physical separation as practicable. The purpose of this separation is to prevent total loss of the hydraulic systems in the event of a malfunction such as fire, or rotor burst wherein one projectile could disable both systems.

(5) Dual actuators must be designed to assure that any single failure, such as a cracked housing, broken interconnecting input, or output link, does not result in loss of total hydraulic system function.

(6) If the assumption under (b) above does not apply and the pilot can control the rotorcraft without undue fatigue after loss of the hydraulic system, then a single hydraulic system is acceptable.

(7) The pressure-indicating system required by § 29.1435, paragraph (a)(3), can be satisfied with a dial, vertical scale, or digital indicator. The indicator should enable the crew to detect pressure trends. Paragraph AC 29.1322 concerns § 29.1322 regarding proper colors for annunciators if they are used to supplement the indicating system.

(8) An analysis or a combination of analysis and tests must be included in the substantiating data file to show compliance with paragraphs (a)(1), (a)(2), and (a)(4) of § 29.1435.

(9) Extra caution should be exercised to assure that control input forces at the mechanical connection to the actuator pilot valves do not exceed their intended value. Consideration should be given to the most adverse tolerance buildup in parts fabrication and control system rigging.

(10) The substantiating data should show that the hydraulic components will perform their intended function reliably under the most adverse continuous and short-time environmental conditions to which they are exposed. These variables include but are not limited to temperature, humidity, vibration, altitude, and shock. Paragraph AC 29.1309b(2)(i) is a method of temperature correction to cover the entire operating temperature envelope being certified.

(11) The system component strength must be sufficient for its material fatigue life to exceed the number of cycles imposed by pump ripple pressure.

c. Installation Precautions and Fire Protection.

(1) All components and tubing routed through fire zones may be designed to comply with the fire protection requirements of §§ 29.1183, 29.1185, and 29.1189. As an alternative, a fireproof shield may be used around the component to be protected. The component should be sufficiently protected to assure fluid leakage will not occur and fuel the fire.

(2) All hydraulic lines should be sufficiently isolated from the engine bleed air lines, environmental control unit, oil cooler, or other heat source to assure expected line life.

(3) If flammable hydraulic fluid is used, the hydraulic components should be isolated from ignition sources to assure that failure of any of the hydraulic components will not result in a fire or explosion. In the case of electrical ignition sources in the proximity of hydraulic components, the electrical equipment should be hermetically sealed or otherwise substantiated as not being an ignition source. (Reference paragraph AC 29.1309b(1)(i).)

(4) The installation detail should be thoroughly reviewed for adequacy of line clamping and clearance from sharp edges. As much physical separation as possible should be provided between hydraulic lines and electrical cables.

(5) While the control system is being moved from stop to stop, observation should be made to determine that hose flexing and tube bending is minimized.

d. Testing.

(1) Individual components should be substantiated by either vendor's or primary manufacturer's laboratory test reports. These tests should establish performance ratings such as pressures, flow rates, environmental capability, etc., to be approved.

(2) After the total system is installed, ground tests should be conducted to assure the system performs as intended and that each component is functioning within its design rating.

(3) If the total system design permits each combined independent power source and actuator to be disabled by shutoff valves, engine shutdown, etc., each combination should be disabled and the remaining combination verified to perform the necessary control functions. The test should be accomplished again with the functioning combination disabled and the disabled combination functioning. These tests should be accomplished first by ground tests, then repeated in flight.

(4) Temperature and pressure instrumentation should be provided at the critical points in the system to meet the provisions of d(2) above. Temperature results should be corrected for hot day conditions. (Paragraph AC 29.1309b(2)(i) gives a recommended procedure.)

(5) All controls should be cycled throughout their complete range of travel while accomplishing d(2) above.

(6) Satisfactory hydraulic system performance should be verified while the pump drive sources (rotor, engine, etc.) are individually varied throughout their approved operating range.

(7) Flight tests should be conducted throughout all altitudes, maneuvers, and control ranges while the system is instrumented as in d(2) and (4) above to determine that component ratings are not exceeded.

AC 29.1439. § 29.1439 PROTECTIVE BREATHING EQUIPMENT.

a. Explanation. This paragraph prescribes minimum requirements for eye and respiratory protection from toxic atmospheres during in-flight emergencies if one or more cargo or baggage compartments are to be accessible in flight. The equipment provided shall assure the crew protection against an oxygen deficient, toxic or highly irritating environment such as smoke.

b. Procedures.

(1) The equipment should provide a good fit for the range of intended users.

(2) A donning procedure should be provided by the manufacturer, evaluated, and the final procedure included in the Rotorcraft Flight Manual.

(3) The equipment should accommodate crewmembers who wear corrective glasses. Nominal position of eyeglasses should not be compromised. The equipment should not cause distortion or undue discomfort.

(4) The equipment donned under the stress of emergency should orient to the face and head, and interface to mating equipment, if required, in an obvious and

uncomplicated manner. Respiratory and eye protection should be provided in a manner that does not compromise the crew's ability to perform required tasks.

(5) Any system that interfaces with existing components, should demonstrate satisfactory performance when operated with these components.

(6) For systems that require positive pressure to furnish satisfactory protection, a positive pressure vs. gas consumption curve should be supplied with the system along with instructions on the proper matching of the system or components to assure the minimum duration requirements of the standard are met.

(7) TSO-C99 and C116 are for Protective Breathing Equipment. If equipment is considered that is not qualified to one of these TSO's, it is recommended that their provisions be reviewed and used as a basis for a qualification program for the equipment being considered. TSO-C99 provides minimum performance requirements for emergency equipment which provides flight deck and cabin crewmembers with eye and respiratory protection from toxic atmospheres during in-flight emergencies. TSO-C116 results in protective breathing equipment that provides any crewmember with the ability to locate and combat a fire within the aircraft cabin or any other accessible compartment.

(8) Additional information regarding oxygen supply systems can be found in paragraph AC 29 MG 6.

AC 29.1457. § 29.1457 (Amendment 29-6) COCKPIT VOICE RECORDER.

a. Explanation. The function of the cockpit voice recorder (CVR) is to provide a record of the crew communications preceding an accidental crash of the rotorcraft. Over the last several years, the National Transportation Safety Board (NTSB) has determined that CVR's are invaluable in determining probable cause of an accident. Because of this fact and acts of Congress, the use of CVR's is required on many rotorcraft involved in passenger-carrying operations.

b. Procedures. The following areas are of particular consideration in the approval of a CVR installation.

(1) Equipment Qualifications. The CVR must be approved. The most common way of obtaining an approval is to qualify the CVR (and associated control panel, if appropriate) to TSO C84.

(2) Cockpit Area Microphone (CAM). The third channel of recorded information is specified to be from a cockpit area microphone or from voice activated lip microphones at the first and second pilot stations. It should be noted that a continuously recording or "hot" microphone at both the first and second pilot stations would satisfy this CAM requirement. Due to the ambient noise level in rotorcraft, the use of "hot" microphone results in objectionable constant "hissing" in the pilot's

headsets. Therefore, it is recommended that “hot” microphones not be used on rotorcraft.

(3) CVR Mechanical Installation. The CVR or the portion thereof which contains the recording should be physically located to enhance the probability of the recording surviving a crash. Normally, such a location would be in the lower portion of the rotorcraft as far aft as possible.

(4) Intelligibility of Recordings. Tests should be accomplished to determine that the recording is intelligible enough to make a positive identification of the speaker and the words or phrases spoken. This is usually accomplished by a flight test that provides an operation to produce the maximum cockpit background noise. The operation should provide for the normal speech of all crewmembers to be recorded on the pertinent channels. Then, during playback, preferably using a different listener, the listener should be able to identify the different crewmembers, the words and phrases spoken by the crew, and the radio communications made by and to the crew. The use of special filters and multiple playbacks to improve intelligibility is acceptable.

(5) Electrical Power Supply. The rule requires that the CVR should be supplied with power from a reliable source that does not jeopardize essential or emergency loads. For Category A rotorcraft, the CVR is not an essential load as specified in § 29.1309(e). However, since the functioning of the CVR is required by operating rules for some operations, it should be given priority over other nonessential loads.

(6) Self-Test Function. The CVR should be provided with a means in the cockpit that will allow a test to ensure the CVR is functioning properly. This may be accomplished by a manual playback feature.

(7) Bulk erasure. If this function is provided, the installation should be as follows:

(i) Any probable malfunction will not cause erasure of the recording medium.

(ii) The crash impact forces will not cause activation of the bulk erasure function.

(iii) Inadvertent actuation of the bulk erasure function is minimized. Usually, this is accomplished by requiring two separate actions to operate the bulk erasure.

AC 29.1459. § 29.1459 (Amendment 29-25) FLIGHT RECORDERS.

a. Explanation. The function of the flight recorder, sometimes referred to as a flight data recorder, is to provide a record of various aircraft and air data parameters during the operation of the rotorcraft. This data is utilized by accident investigators to aid in determination of the probable cause of an accident. The problems associated with acquisition of this data in aircraft not equipped with flight recorders has been complicated by the use of advanced instrument systems such as EFIS, EICAS, and IDS. The very nature of the operation of these systems precludes the deduction of post-accident data, as was possible with mechanical and electromechanical instruments, annunciators and switches. The National Transportation Safety Board (NTSB) therefore made a recommendation to the FAA that aircraft should be required to have flight recorders. Subsequently Congress mandated that flight recorders be required on many rotorcraft involved in passenger-carrying operations in accordance with FAR 91 and FAR 135.

b. Procedures. The following areas are of particular consideration in the approval of a flight data recorder installation.

(1) Equipment Qualification. The recommended procedure to obtain an approval for the flight recorder (and associated control panel, if appropriate) is to qualify the flight recorder to TSO C-124. The required underwater locating device should be qualified to the provisions of TSO C-121.

(2) Recorded Parameters and Accuracy.

(i) Airspeed. The installed flight recorder for a Category A rotorcraft should record the airspeed with an accuracy of 3 percent or 5 knots (whichever is greater) from a speed of 80 percent of V_{TOSS} to V_{NE} in level flight, and an accuracy of 10 knots from a speed 10 knots less than V_{TOSS} to a speed of 10 knots more than V_Y in climb.

(ii) Pressure Altitude. The flight recorder should be capable of recording the pressure altitude of the rotorcraft with a range of -1,000 feet to the maximum certified altitude. The error of this recording at sea level, excluding instrument calibration error, should not exceed ±30 feet or a value of ±30 feet for each 100 knots of airspeed (whichever is greater).

(iii) Direction. The flight recorder should be capable of recording the magnetic heading of the rotorcraft within at least 10 degrees for any heading.

(iv) Vertical Acceleration. The flight recorder should be capable of recording the normal acceleration within the center of gravity range of the rotorcraft. The recommended range of this recording is an envelope of -3 to +6 G with an accuracy of at least ±0.2 G.

(v) Time Correlation. The flight recorder should provide a time scaled correlation between the data recorded and the time at which this information was presented to the first pilot via his required flight instruments. This correlation should normally be established before flight, and should have an accuracy rate that does not diverge by more than 4 minutes and 4 seconds in 8 hours.

(vi) Caveat. It should be noted that even though the requirements outlined above provide for compliance with the specific provisions of § 29.1459 regarding the acquired data and its accuracy, a flight recorder certified to these minimum standards will not meet the requirements of Appendix F of FAR 91 or Appendix C of FAR 135. If the flight recorder is to be used to comply with these operating rules, it is recommended that the appropriate appendix be consulted prior to requesting certification. The approved configuration may then be certified as meeting the requirements of the appropriate appendix.

(3) Flight Recorder Mechanical Installation. The non-ejectable flight recorder or the portion thereof which contains the recorded data should be physically located to enhance the probability of the recording surviving a crash. Normally, such a location would be in the lower portion of the rotorcraft as far aft as possible. However other locations in the rotorcraft may be suitable to meet the requirement to “minimize the probability of container rupture resulting from crash impact and subsequent damage to the record from fire.” The normal accelerometer should be located within the most restrictive center of gravity of the rotorcraft. The required underwater locator is usually mounted to the case of the flight recorder.

(4) Electrical Power Supply. The rule requires that the flight recorder should be supplied with power from a reliable source that does not jeopardize essential or emergency loads. For Category A rotorcraft, the flight recorder is not an essential load as specified in § 29.1309(e). However, since the functioning of the flight recorder is required by operating rules for some operations, it should be given priority over other nonessential loads.

(5) Self-Test Function. The flight recorder should be provided with a preflight test which will provide confirmation that the recorder and its recording medium are functioning properly.

(6) Data Erasure Feature. If this function is provided and the flight recorder is not powered solely by an engine or transmission driven generator, the installation should provide the following features:

(i) Any probable malfunction will not cause erasure of the recording medium.

(ii) The crash impact forces will not cause activation of the data erasure function.

(iii) Inadvertent actuation of the data erasure function is minimized. Usually, this is accomplished by requiring two separate actions to operate the data erasure.

AC 29.1461. § 29.1461 (Amendment 29-3) EQUIPMENT CONTAINING HIGH ENERGY ROTORS.

a. Explanation. This section contains requirements for the installation of equipment containing high energy rotors. A high energy rotor is any rotor which has sufficient kinetic energy to cause damage to surrounding structure, wiring, and equipment if a failure occurs. Turboshaft engine and APU rotors are not covered by this paragraph. One of the following requirements of § 29.1461 must be met.

(1) Paragraph (b) deals with damage tolerance, containment, and control devices.

(2) Paragraph (c) deals with containment and inoperative speed controls.

(3) Paragraph (d) deals primarily with equipment location.

b. Procedures.

(1) Compliance with § 29.1461(b) can be shown by a combination of analysis and test. A failure modes and effects and a stress analysis, together with a dynamic test, could be used to verify that the rotor would withstand the damage from environmental effects, and that the rotor case would contain any parts that may separate from the rotor shaft. The analysis and test should include a demonstration of the control device's ability to prevent limitations from being exceeded.

(2) If compliance with the requirements of § 29.1461(c) is chosen, a test must be conducted which demonstrates that all parts from any type failure of a high energy rotor will be contained when that rotor is operating at the highest speed obtainable, with all speed control devices inoperative. This containment must not damage any components, systems, or surrounding structures that are essential for continued safe flight.

(3) If compliance with § 29.1461(d) is chosen, the location of the high energy rotor must be in an area where uncontained failed parts will not damage other components, systems, or surrounding structure which are essential for continued safe flight. It must also be shown that there is no possibility for failed, uncontained parts to enter the cabin area and endanger any occupant.